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Volume II

2 NW

ADVANCED DESIGN COMPOSITE AIRCRAFT (ADCA) STUDY

MANUFACTURING PLAN

GRUMMAN AEROSPACE CORPORATION
BETHPAGE, NEW YORK 11714

NOVEMBER 1976

TECHNICAL REPORT AFFDL-TR-76-97 Volume II
FINAL REPORT FOR JULY 1975 - AUGUST 1976

AIR FORCE FLIGHT DYNAMICS LABORATORIES
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This technical report has been reviewed and is approved for publication.

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COMPOSITE TOOLING
QUALITY CONTROL

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

THE REPORT INCLUDES DETAILED MANUFACTURING PLANS FOR THE FABRICATION OF MAJOR STRUCTURAL COMPONENTS OF THE ADCA AIRCRAFT AND THEIR FINAL ASSEMBLY. A SURVEY OF EXISTING TOOLING/MANUFACTURING TECHNOLOGY IS PRESENTED TO HIGHLIGHT AREAS REQUIRING FURTHER DEVELOPMENT.

A CORROSION-CONTROL PLAN AND QUALITY-CONTROL PLAN ARE PRESENTED TO SHOW THEIR RELEVANCE TO THE MANUFACTURING PLAN.

BASED ON A COMPARISON TO CURRENT METAL AIRCRAFT, THE DESIGN/MANUFACTURING PHILOSOPHY WAS ESTABLISHED TO REDUCE COMPOSITE PROGRAM COSTS, BY STRIVING TO REDUCE PART/FASTENER COUNT.

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FOREWORD

The work reported herein was performed under the sponsorship of the Air Force Flight Dynamics Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio 45433. Mr. L. Kelly, AFFDL/FBS is the Air Force Program Manager and Cpt. E. Bannink, AFFDL/FBS is the Air Force Project Engineer.

The work was performed by the Advanced Composites/Advanced Aircraft Systems Groups, Grumman Aerospace Corporation, Bethpage, New York 17714, under Contract F-33615-75-C-3124, Project Task 1368-01-31. The inclusive dates of contract performance are July 1975 to Aug. 30, 1976. Submittal Date of this report is November 1976.

Key Grumman personnel associated with the program and their respective areas of responsibility are:

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S. Johnson	Structural Analysis	T. Ervolina	Propulsion
C. Waters	Dynamics	C. Gianetto	Guidance & Control
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D. Sharp	Structural Mechanics	D. Colquhoun	Configuration Design

J. Katz — Engineering Specifications

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Section I
INTRODUCTION AND SUMMARY

The ADCA aircraft design study was conducted using a design-manufacturing interface to develop a producible structure which met all design requirements. Emphasis was placed on low vehicle acquisition cost in accordance with current budgetary restrictions. Therefore, this study began with a review of the high cost centers in the fabrication of current high-performance metal aircraft. Although metalworking technology is highly advanced, production costs are high due to the extremely large number of complex detail parts which must be produced and delivered on schedule to the assembly areas. The assembly flow is further complicated by the installation of countless fasteners as the detail parts are assembled into modules which, in turn, are mated into the vehicle's basic structural components. Additional costs accrue from the fixturing, mechanical fastening equipment, and factory space required for assembly operations.

Therefore, a design-manufacturing philosophy for the composite ADCA vehicle was developed which significantly reduced part and fastener count. Wherever possible, cocuring of large assemblies was used to reduce part count, thereby lowering tooling, fastening, and production tracking costs. The currently available composite fabrication and tooling technology was reviewed to assess areas requiring further improvement. Three new concepts were developed. The translaminar reinforcement (TLR) concept uses stitching through the laminates to eliminate the anti-peel fasteners required for integrally molded skin-to-angle attachments. Thus, the cocured Gr/Ep assembly provides for joining skin to web components using significantly fewer fasteners and detail parts with lower-cost tooling. Also, the conformal molding process was developed so that mating parts, which must be subsequently fastened, are molded concurrently to eliminate fit-up problems. The punched fiber reinforcement (PFR) joint design places graphite yarns inserted through the skin into the bondline of the webs of back-to-back channels bonded to the cover. The resulting joint transmits shear load through the bond and fuel pressure load as tension in the yarns. These three concepts are used in fabricating the wing box.

Numerous sandwich panels are used in the aircraft; the majority of these will be manufactured by cocuring rather than costly secondary bonding. The cocuring process is also used to integrally mold surface protection systems for lightning strike, foreign object impact, and damage during maintenance to further simplify fabrication flow. Secondary bonding is limited to the main wing spars, canard, and vertical fin torque box due to various factors such as sequencing, inspection constraints, and tooling limitations.

The major structural attachment fittings are titanium/steel machined parts due to the triaxial state of stress and high attachment loads. Such type components were the only limitation on composite utilization in the aircraft.

Manufacturing plans were prepared for fabricating the wing, vertical and dorsal fins, rudder, canard, and fuselage assemblies. Each plan includes a tooling concept, an evaluation of various production schemes, and a description of the manufacturing sequence for the process selected. A final assembly plan was generated to cover joining of the various components. Sealing and finishing requirements were established. A quality control plan was also prepared to provide confidence in the structure's inspectability.

The fabrication sequences considered for the ADCA aircraft are all feasible, cost-effective, and minimize manufacturing risk. With available technology, the vehicle will cost less than its metal counterpart due to reductions in part and fastener count. In the B-1 Composite Horizontal Stabilizer program, a similar approach led to projected cost savings of 17.5%. Future savings will accrue as the differential between composite and metal raw materials cost is reduced, and composite material forms with increased processibility are introduced. Innovative manufacturing techniques under current development will also simplify fabrication flow and thus also reduce cost.

In summary, fabrication of an essentially all-composite aircraft has been shown to be both feasible and cost-effective. Manufacturing process confirmation would be obtained during fabrication of design verification components so that the full-scale aircraft can be produced with confidence.

Section II
ADVANCED DESIGN COMPOSITE AIRCRAFT STRUCTURAL ARRANGEMENT

The ADCA aircraft configuration utilizes a trisonic hybrid wing planform with close-coupled canards and an afterburning F-101-GE-100 engine as shown in Figures 1 and 2. The fuselage contains three main modules with a structural engine duct. Extensive use is made of honeycomb and shell liner-stiffened panels in the fuselage. The longerons are integrally molded into the skin and spliced at manufacturing breaks. The wing centerbox is of spar and rib construction and extends from tip to tip. The canards and vertical fin are of sandwich construction.

Honeycomb sandwich panels are used for the leading and trailing edges, and tip caps for the wings, vertical fin, and canard. The major structural attachment fittings are titanium machined parts or weldments mechanically attached to the composite structures.

Graphite/Epoxy is the primary structural material with boron/graphite/epoxy, Kevlar/graphite/epoxy hybrids, and glass/epoxy laminates used locally. Graphite/epoxy and glass/epoxy are the preferred honeycomb materials. Titanium or stainless steel fasteners are used to prevent fastener corrosion. Conventional epoxy film adhesives, epoxy paste adhesives, polysulfide sealants, epoxy/polyamide primers, and linear polyurethane finishes are used as required. Ultrasonically welded, fuel-resistant, thermally stable thermoplastics (polyimide, polyphenylene sulfide, polyarylsulfone) are under consideration for sealing the wing fuel tank. However. the baseline design is shown as a groove channel seal.

DATA

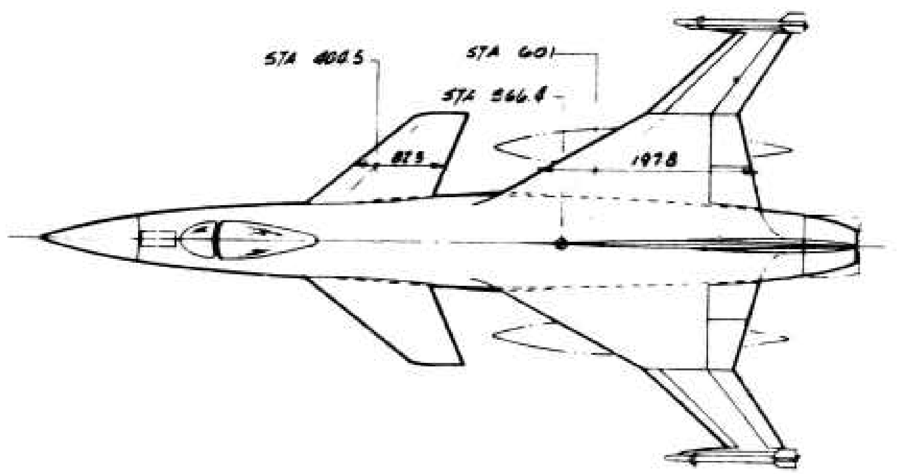
SURFACE	AREA FT²	AR	TR	t/c %	Λ_{LE}
WING	411	2.75	.25/.58	3-5-5	60°/46.5°
CANARD	90	7.19	.4	4-3	50°
VERT TAIL	80	.875	.25	4-3	55°

SURFACE CONTROLS	AREA FT²	MAX DEFLECT	RATE
CANARD	90	±30°	50°/SEC
INBRD FLAPERON	40	±30°	50°/SEC
OUTBRD FLAPERON	20	+20°	30°/SEC
OUTBRD L.E. DROOP	11	20°	30°/SEC
RUDDER	21	±30°	50°/SEC

WEIGHTS	EMPTY	USEFUL LOAD	TOGW	FUEL
	21556 Lbs	18997 Lbs	40553 Lbs	12675 Lbs

PROPULSION	NOZZLE	RATED THRUST (SLS)
F101-GE-100 A/B TURBOFAN	VARIABLE C/D	30 000 Lb CLASS

MAX CROSS-SECTIONAL AREA = 31.9 FT²
TOTAL WETTED AREA = 1931 FT²



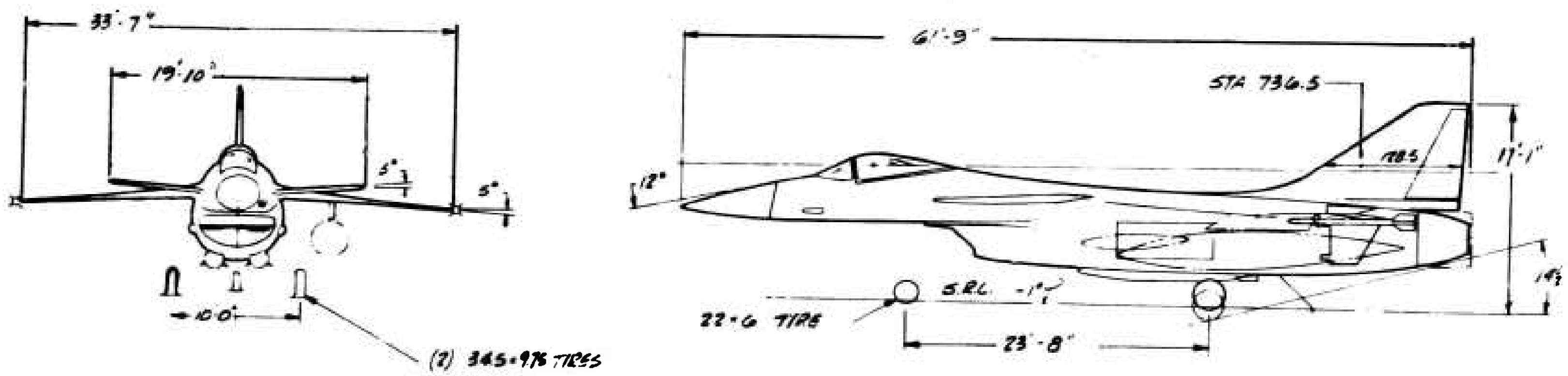


Figure 1. ADCA General Arrangement

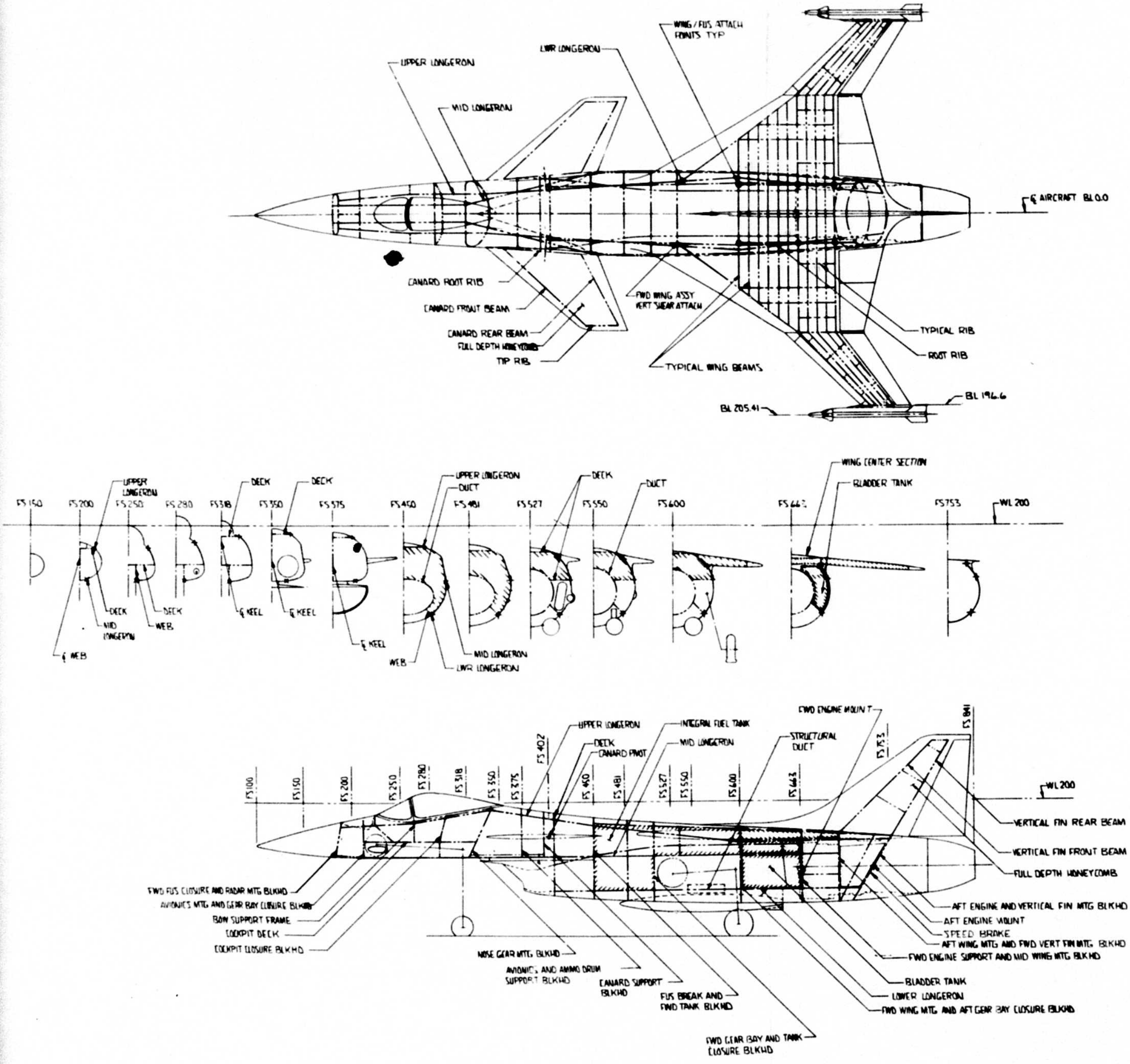


Figure 2. ADCA Structural Arrange

Section III
DEVELOPMENT OF ADVANCED DESIGN COMPOSITE AIRCRAFT MANUFACTURING PHILOSOPHY

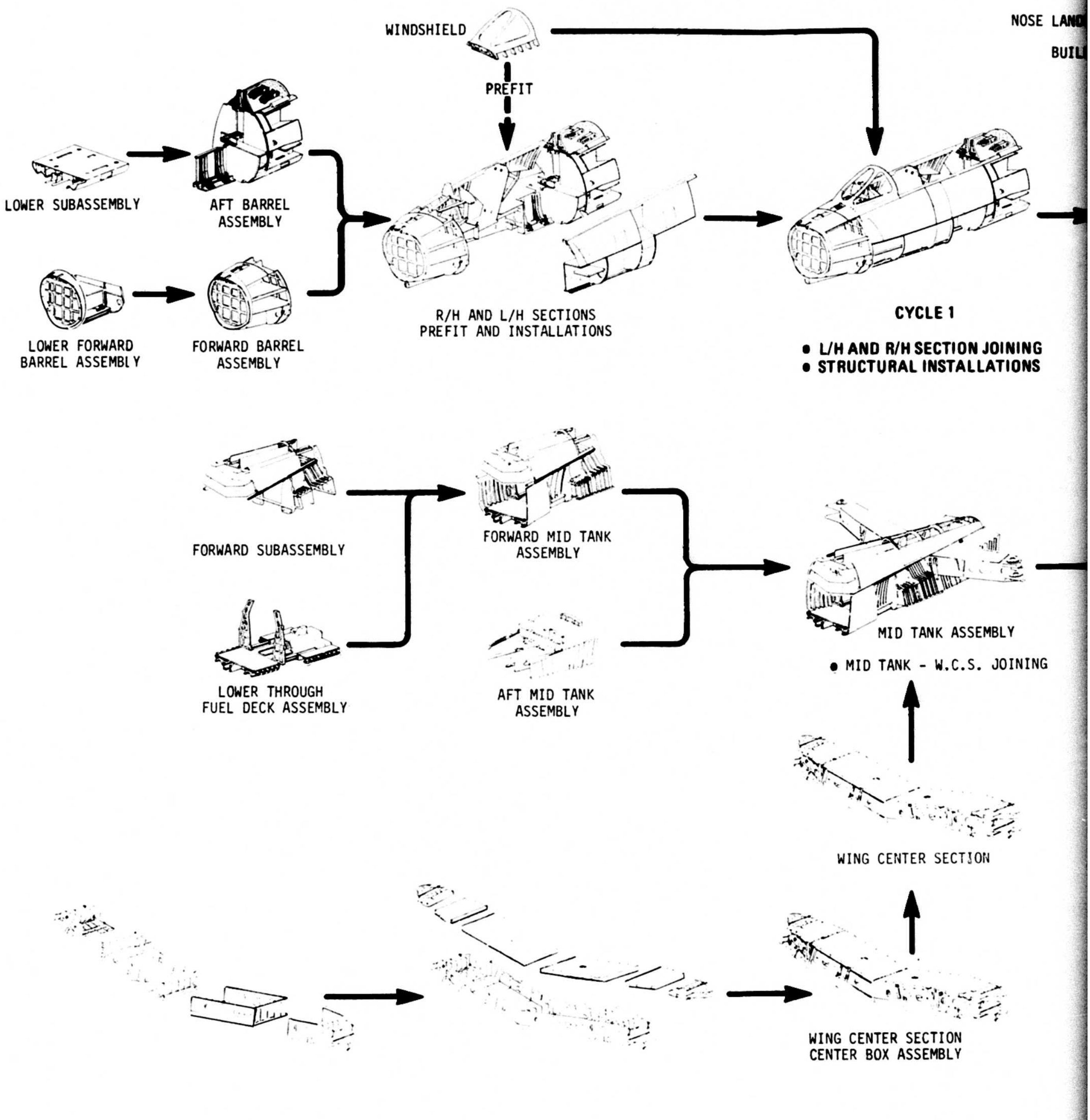
3.1 EVALUATION OF METAL AIRCRAFT FABRICATION

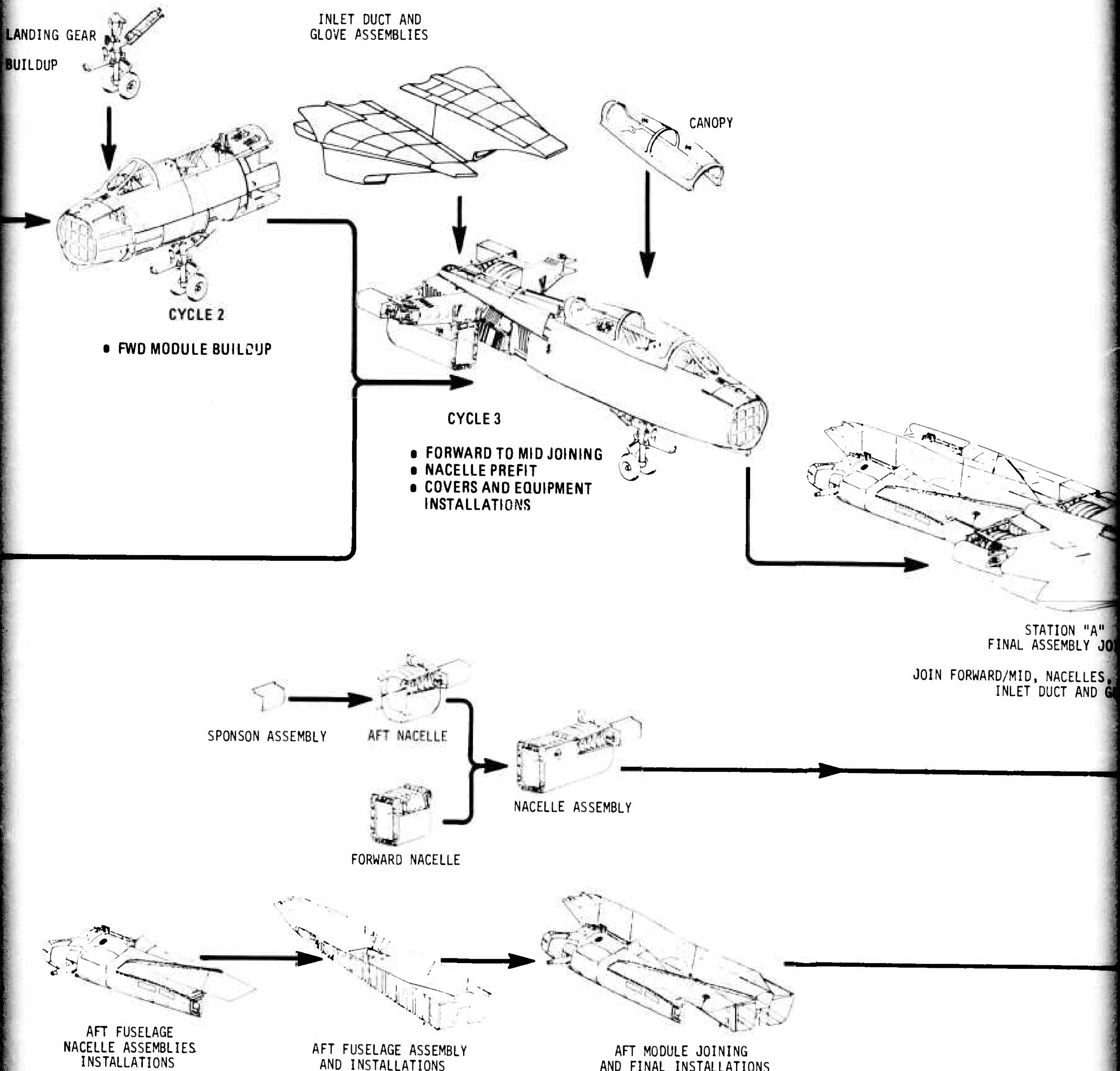
Current metallic advanced fighter aircraft present many manufacturing challenges. These include:

- Large numbers of complex machined and formed parts and bonded assemblies
- Production of intricate, highly loaded weldments
- More stringent drilling and fastener installation requirements to maximize fatigue life
- Accommodating the installation of increasing numbers of systems

The current metal fighter airframes are made of tens of thousands of identifiable detail parts and well over a hundred thousand fasteners. The detail parts are built-up into minor subassemblies which are subsequently joined into major modules. The modules are further assembled into the fuselage, wings, horizontal stabilizers, and vertical fins. These components are ultimately joined to complete the structural assembly process, Figures 3 to 5. For even a moderate production rate, multiple assembly fixtures are needed due to the flow time required to install fasteners despite increasing use of automated machines. In addition to the capital investment required, up to 20% of the production force is estimated to be involved in drilling/fastener installation operations. To further complicate metal aircraft assembly, detail parts from hundreds of sources must arrive in the assembly area as needed to prevent stoppage of the assembly line.

Therefore, manufacturing problems with metal aircraft primarily arise in the large number of detail parts which must be subsequently assembled. Despite advances in detail part fabrication technology, computerized tracking, and automated fastening systems, metal aircraft fabrication flow has not yet been fully controlled.


WINDSHIELD
PREFIT
NOSE LAND
BUIL
LOWER SUBASSEMBLY
AFT BARREL ASSEMBLY
LOWER FORWARD BARREL ASSEMBLY
FORWARD BARREL ASSEMBLY
R/H AND L/H SECTIONS PREFIT AND INSTALLATIONS
CYCLE 1
• L/H AND R/H SECTION JOINING
• STRUCTURAL INSTALLATIONS
FORWARD SUBASSEMBLY
FORWARD MID TANK ASSEMBLY
LOWER THROUGH FUEL DECK ASSEMBLY
AFT MID TANK ASSEMBLY
MID TANK ASSEMBLY
• MID TANK - W.C.S. JOINING
WING CENTER SECTION
WING CENTER SECTION CENTER BOX ASSEMBLY


LANDING GEAR
BUILDUP
INLET DUCT AND
GLOVE ASSEMBLIES
CANOPY
CYCLE 2
• FWD MODULE BUILDUP
CYCLE 3
• FORWARD TO MID JOINING
• NACELLE PREFIT
• COVERS AND EQUIPMENT
INSTALLATIONS
STATION "A"
FINAL ASSEMBLY
JOIN FORWARD/MID, NACELLES,
INLET DUCT AND
SPONSON ASSEMBLY
AFT NACELLE
NACELLE ASSEMBLY
FORWARD NACELLE
AFT FUSELAGE
NACELLE ASSEMBLIES
INSTALLATIONS
AFT FUSELAGE ASSEMBLY
AND INSTALLATIONS
AFT MODULE JOINING
AND FINAL INSTALLATIONS

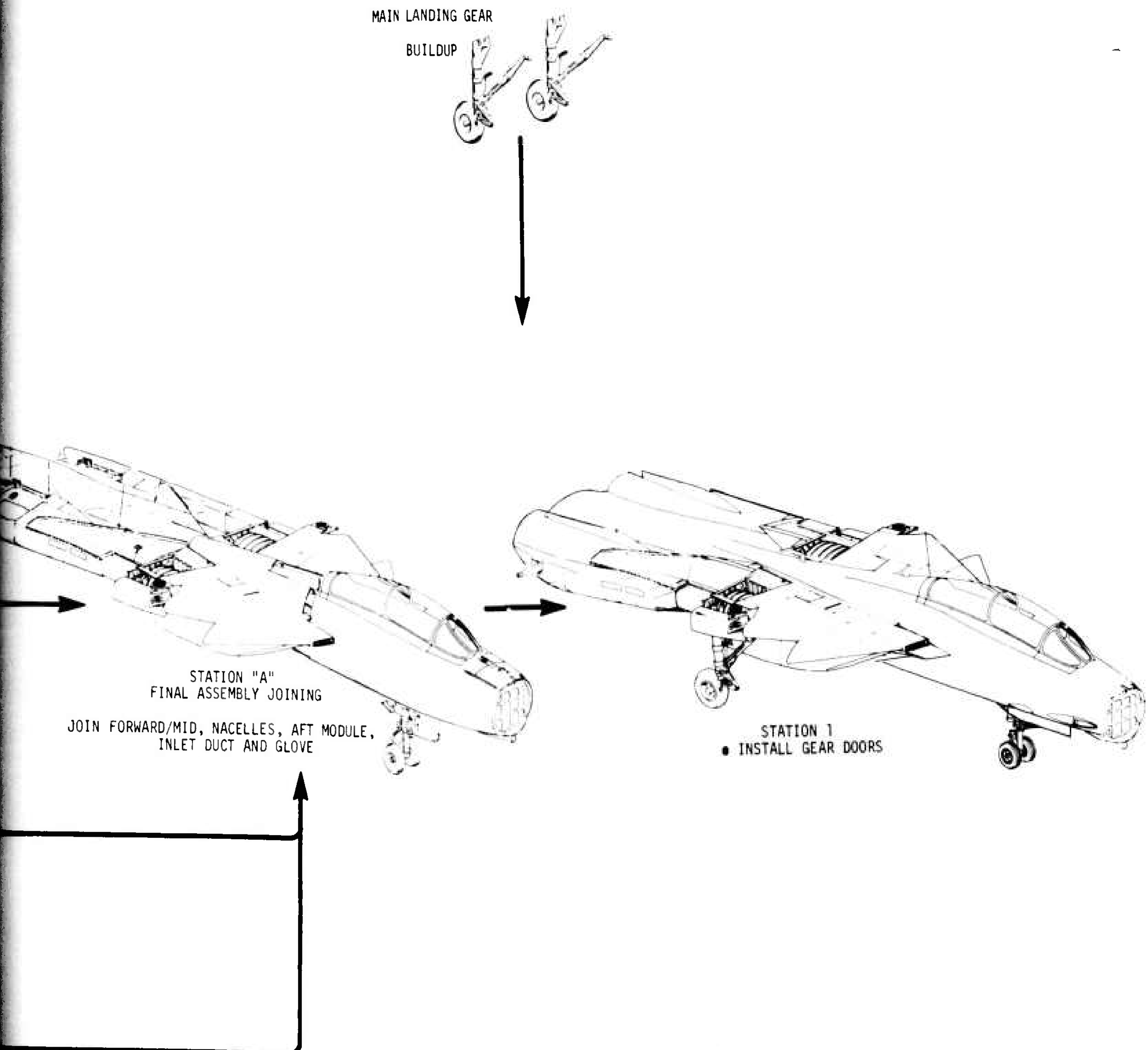


Figure 3. F-14A Fuselage Assembly Flow

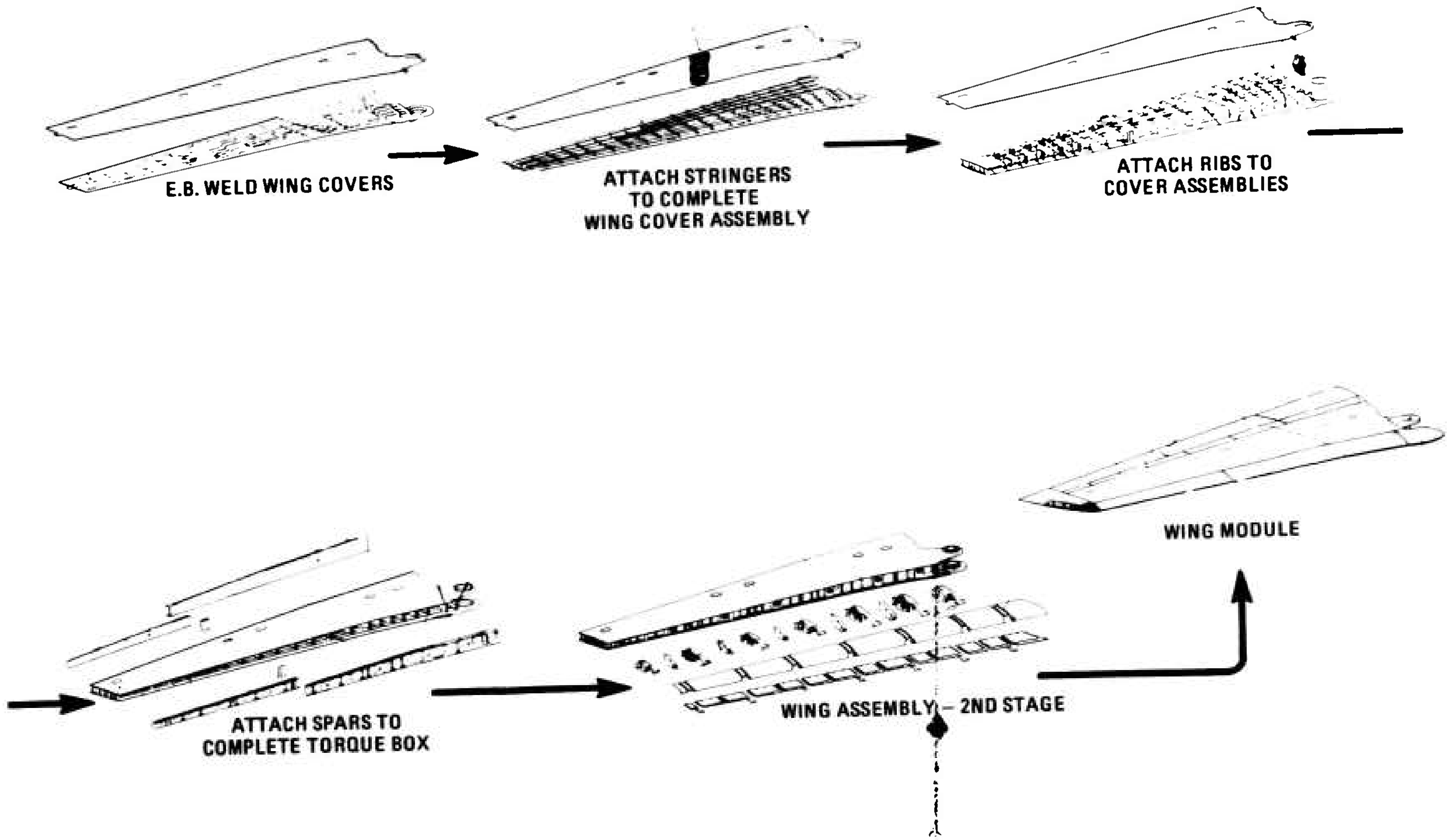
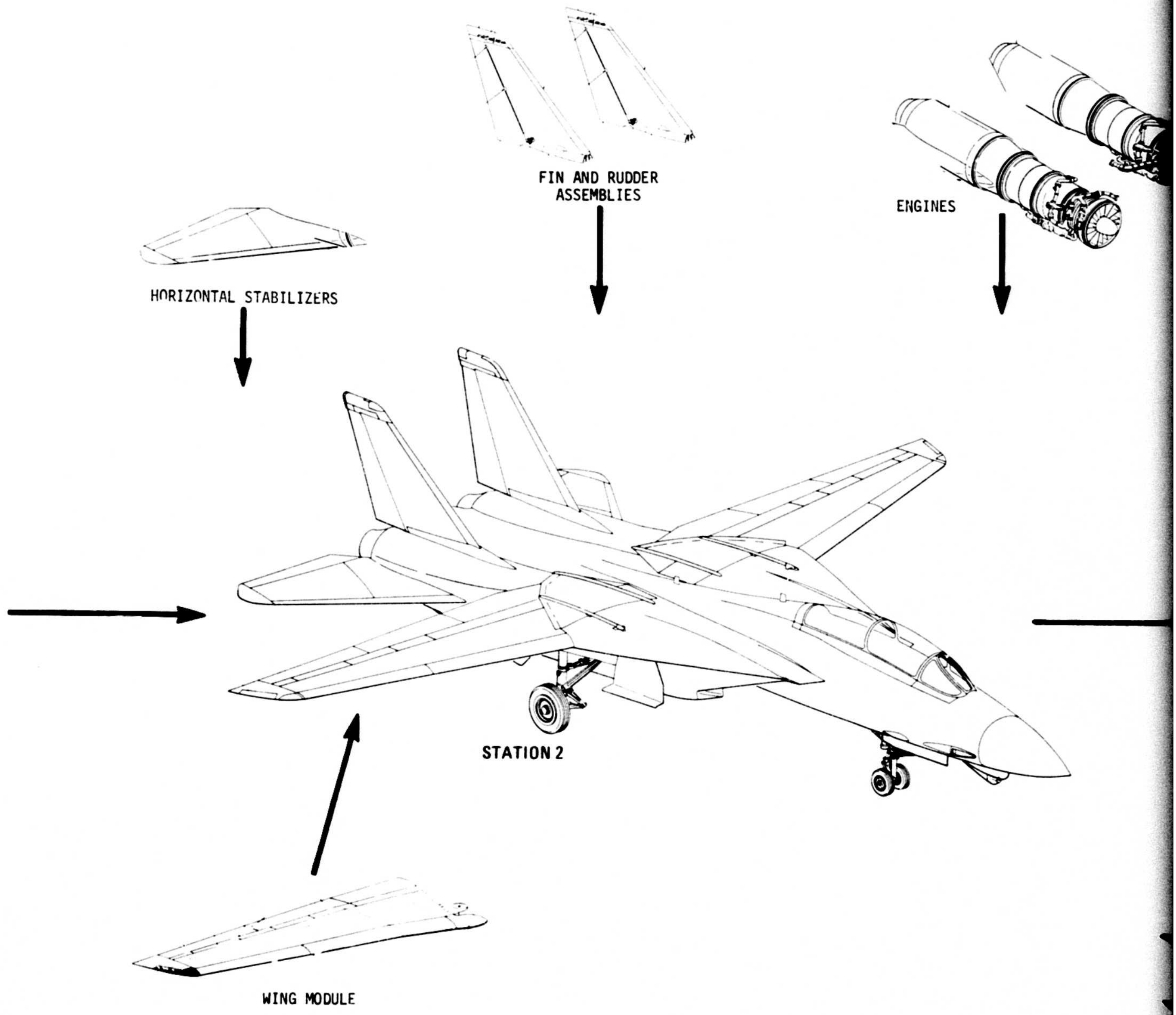


Figure 4. F-14A Wing Assembly Flow


FIN AND RUDDER ASSEMBLIES
ENGINES
HORIZONTAL STABILIZERS
STATION 2
WING MODULE

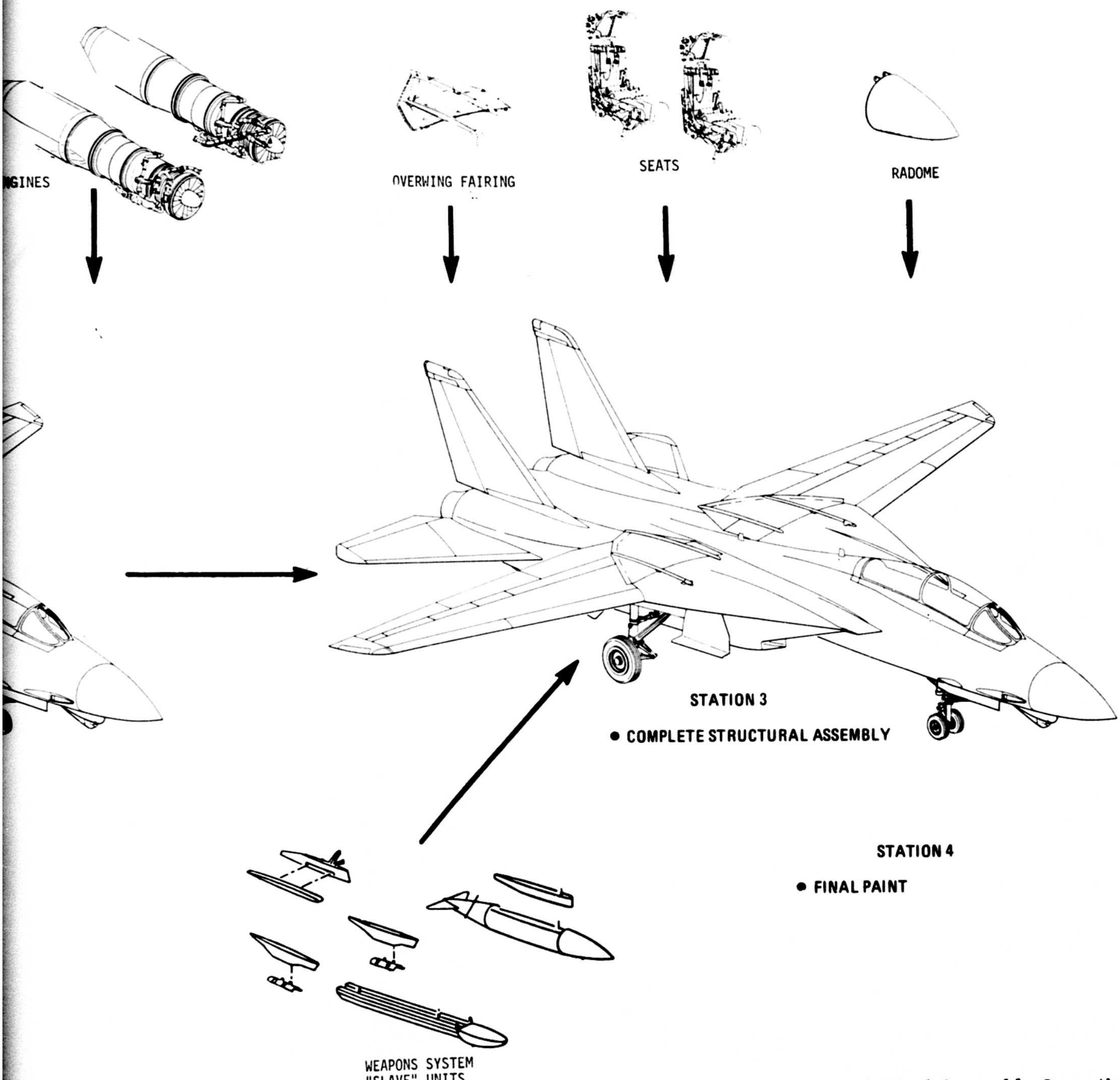


Figure 5. F-14A Final Assembly Operations

3.2 COST-EFFECTIVE COMPOSITE MANUFACTURING PHILOSOPHY

The current concept of advanced composite manufacturing recognizes the problems encountered with metal construction and minimizes their impact on schedule and cost. An active design-manufacturing-quality interface must be maintained to enhance the structure's producibility by reducing part and fastener count to simplify part/assembly flow, Figures 6 and 7. The result of this interface on the ADCA program is a design which includes:

- Molding large sections to minimize production breaks
- Cocuring of skin assemblies with integrally molded internal attachment points and stiffeners, or sandwich panels
- Using integral-molded flanges rather than clips
- Simplifying the core in sandwich structure
- Using joint designs requiring one row of fasteners whenever possible
- Reducing fit-up and sealing problems
- Minimizing types of materials and fasteners required

The proposed manufacturing plan discussed in the following sections has a driver to further expedite part/assembly flow by making use of advanced manufacturing technology. The key points include:

- Designing combination tools or low-cost tools derived from another, more expensive tool made for another purpose
- Using automated layup and drilling machinery
- Molding-in pickup points to coordinate one part to another
- Centralizing work and inspection areas to simplify tracking and scheduling

This detailed plan has been reflected in the ADCA design, yielding an aircraft that is lower in cost than a similar metallic vehicle. The composite manufacturing philosophy has already been verified by the B-1 Horizontal Stabilizer and the F-14A Over-wing Fairing and Main Landing Gear Door programs.

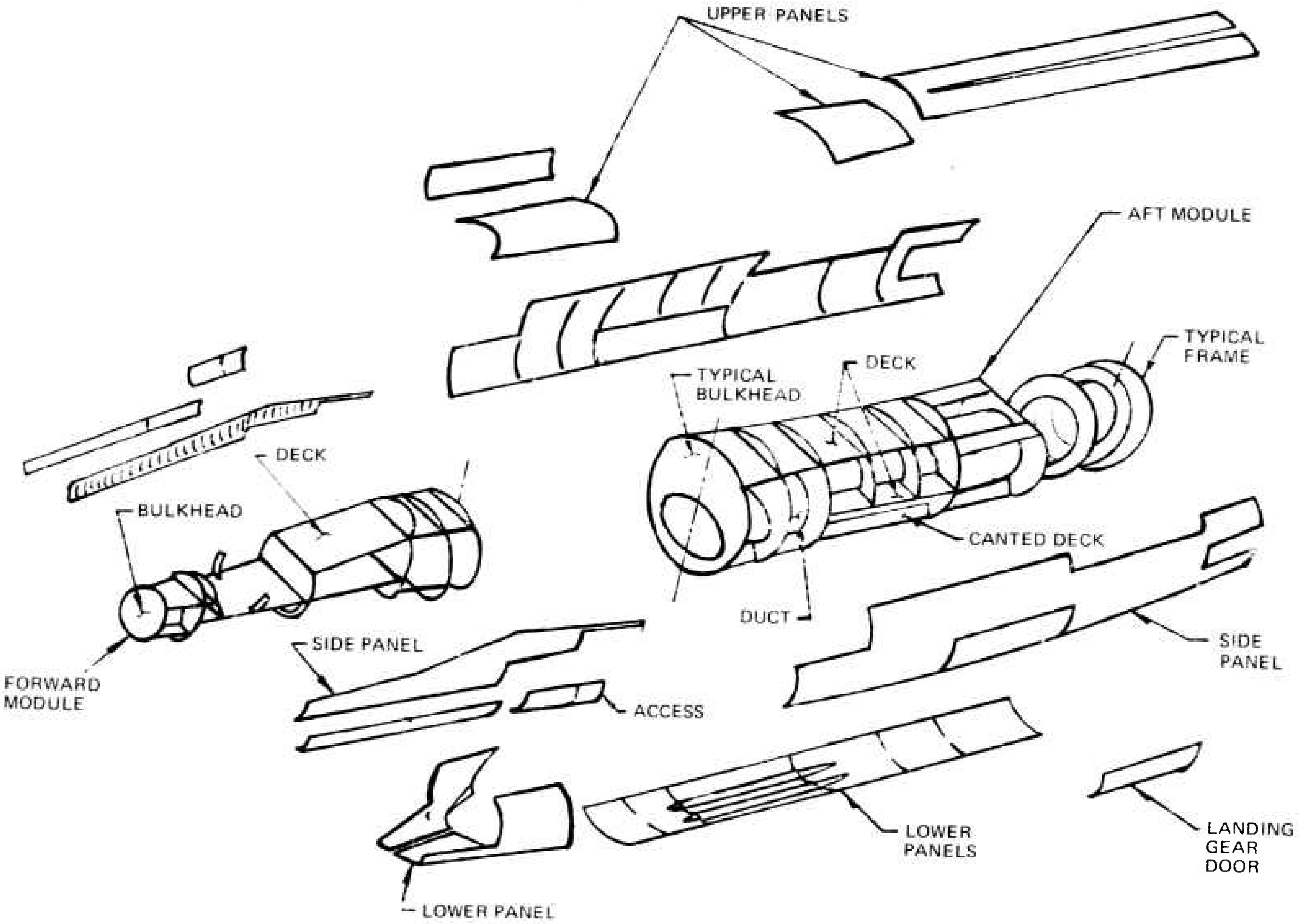


Figure 6. Major Assemblies–Composite Fuselage

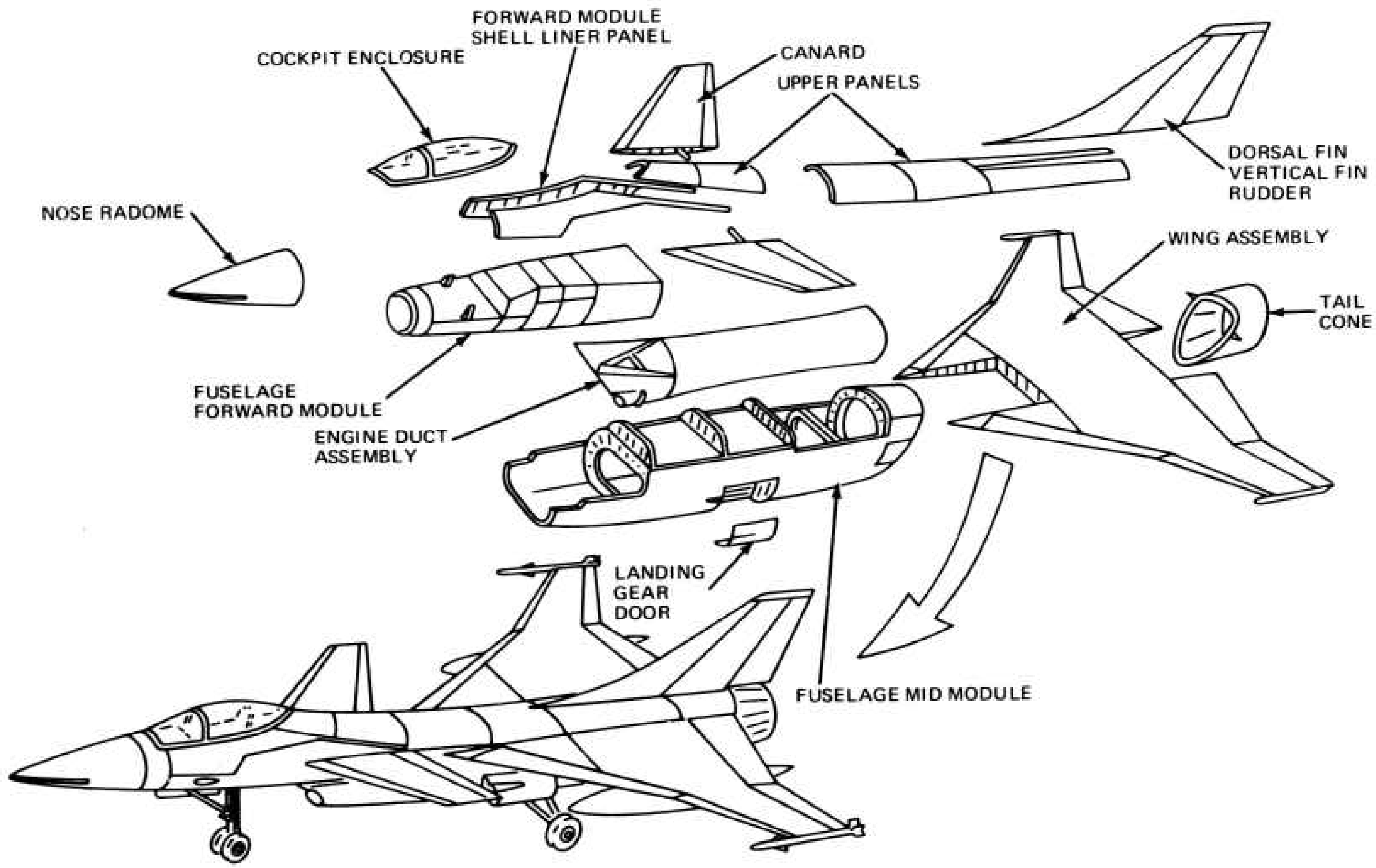


Figure 7. Major Assemblies-ADCA Aircraft

Section IV
APPLICABLE COMPOSITE MANUFACTURING TECHNOLOGY

The manufacturing technology for advanced composite structures has recently undergone a major evolution to produce larger structures and reduce costs. This technology base includes advances in tooling, mechanized layup, automated drilling and cutting, and cocuring of bonded structure. This new technology is capable of further growth to accommodate specific ADCA aircraft requirements.

4.1 DETAIL PART TOOLING

A major development is the verification of dimensionally stable, low-cost mold forms for major fuselage and wing skin sections. These tools are either of Gr/Ep sandwich construction or eggcrate design with relatively thin steel face sheets (Figure 8). High capacity tracer-controlled milling machines recently became available for economically machining

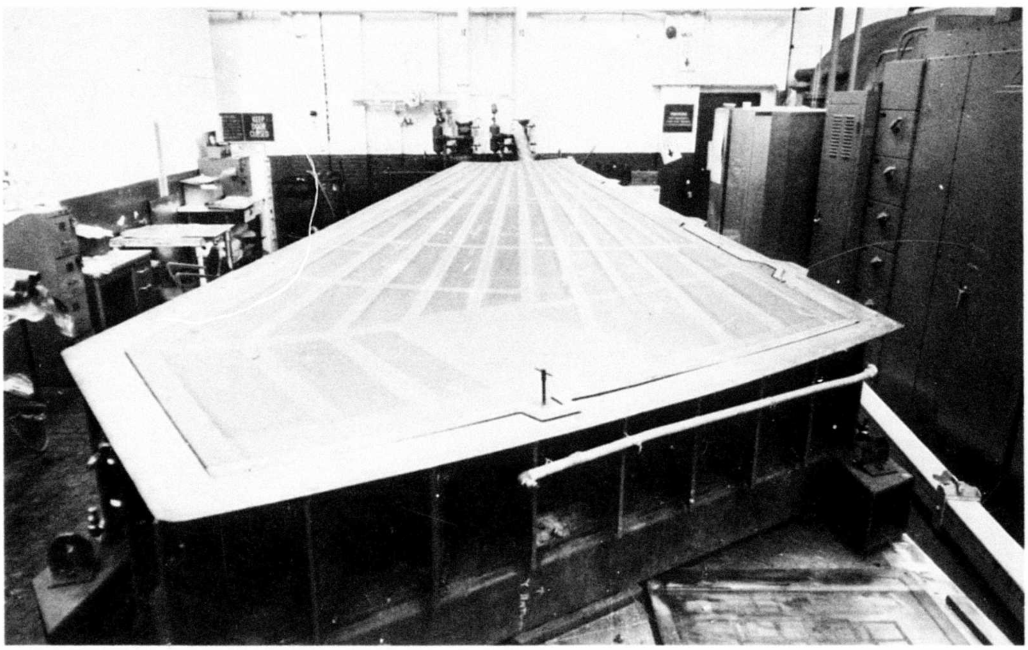


Figure 8. Steel Eggcrate Skin Mold

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contours into solid tools or eggcrate tools (Figure 9). The use of auxiliary tooling coordinated to the mold forms to cocure hat sections and zee stiffeners, integral flanges, and various types of reinforcements and fittings has been demonstrated. New tooling concepts are being developed for complex shape substructure parts due to the inability of rubber tooling to maintain part dimension over production runs. Metal tools, with thin elastomeric faces or deformable materials included in the bleeder system, are currently being verified

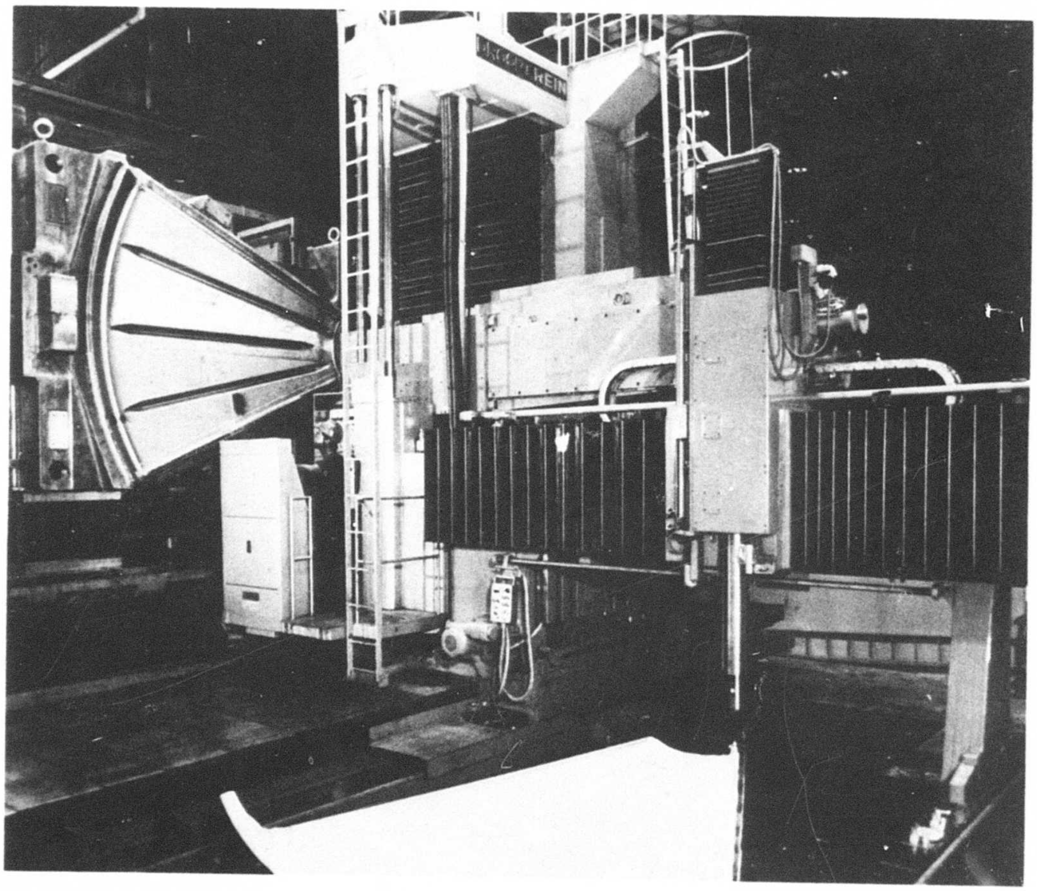


Figure 9. High Capacity Milling Machine

to produce high quality, dimensionally stable parts. The tooling concepts for cocuring various types of skin and airfoil structures have been verified. Molded-in hard points and tooling ears have significantly improved detail parts coordination during mechanical assembly.

4.2 DETAIL PART LAYUP AND TRIMMING

The fabrication of large detail parts has been made more cost-effective by the use of automated layup machinery. The most advanced of these is Grumman's laminating center which consists of three modules: a tape laying head, a N/C-controlled laser for ply trimming, and a stacking device to place the layup in the tool. A TV camera certifies the conformance of the layup to specification requirements. Wider tapes up to 36 in. wide, woven graphite, and preplied sheet materials are available to further reduce layup costs. Mechanical forming devices, such as those used to corrugate the webs of sinewave spars, are available to ensure conformance of the layup to the tool. Improved cure cycles and resin matrices are being verified to enhance the servicibility of Gr/Ep structures for the service life of future aircraft. Controlled flow resin systems are being developed to improve thickness control using low pressure molding processes.

Another area of improved manufacturing technology is ply and laminate trimming. The use of lasers to trim layups of up to three plies thick at rates up to 100 in./min is a production reality. Steel rule die blanking has also been verified. The trimming of thick uncured laminates by water jet and mechanical reciprocating cutters (Gerber Scientific Company) is being developed. Trimming of cured laminates with both fixed and portable machines with diamond grit blades is well established. Water jet cutting of cured laminates is under investigation. Also, increased confidence in net molding detail parts will eliminate the need to trim many parts.

Crossply pultrusion has been demonstrated to be an extremely cost-effective method of producing constant cross-section structural shapes. Minor technology growth is needed to produce localized buildups in highly loaded areas, but this appears feasible by novel design segmented dies to accommodate thickness changes. The fabrication of radically tapered parts appears to be beyond reach of current technology but would provide significant cost savings.

4.3 COCURING PROCESSES

4.3.1 Laminate Assemblies

Despite advances in mechanical assembly, cocure bonding to reduce the required number of parts and fasteners is the more cost-effective manufacturing process. However, for some applications, mechanical attachments must be added to cocured parts as they have low peel resistance. To overcome this problem, Grumman has developed the translaminar reinforcement technique (TLR) in which high-performance fibers such as Kevlar 49 and T-300 are sewn through the substructure and cover prior to cure to form a three-dimensional composite. The stitching improves the joint's peel resistance and reduces or eliminates the need for antipeel fasteners and localized buildups in the skin. During cure, the vertical leg is supported by rubber-faced steel bars and a spacer bar attachment to the mold form. The rubber facing takes up mismatches in the layup and prevents markoff on the skin, while the steel bar provides sufficient rigidity to prevent twisting (Figure 10).

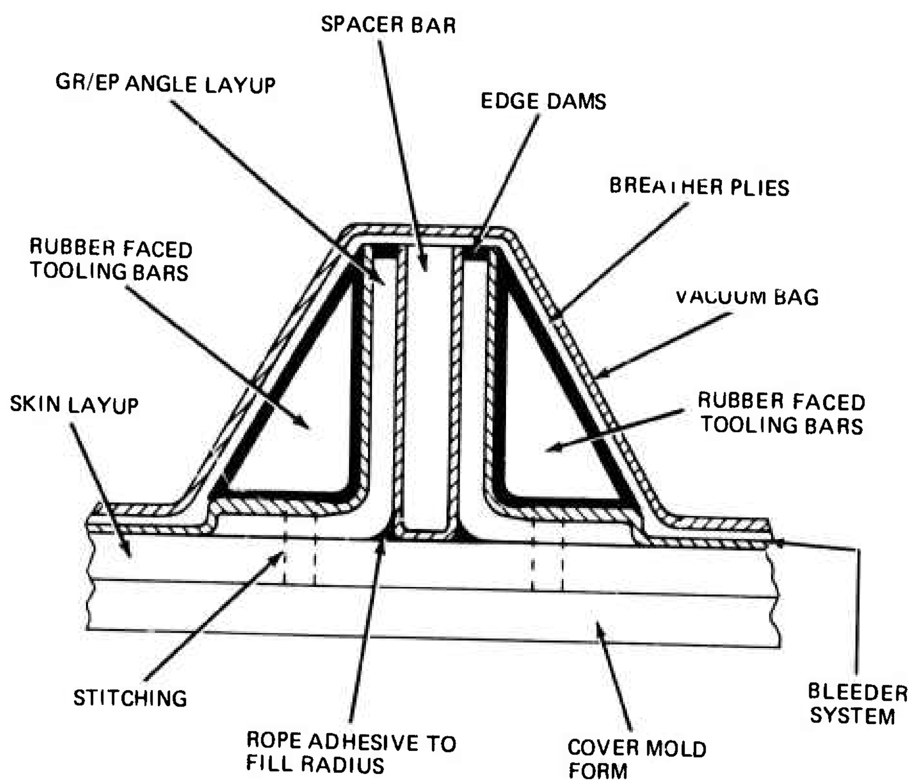


Figure 10. TLR Integrally Molded Attachment Channel

The manufacturing sequence is as follows:

- Layup cover skin in tool
- Layup angle laminates in the flat
- Locate angle laminates over skin, place separator film in areas of vertical leg to prevent sticking
- Stitch laminates together
- Install spacer bars and form vertical legs
- Install bleeder system and rubber-faced tooling bars
- Apply breather plies
- Apply vacuum bag and cure

This cocuring approaeh significantly reduces tooling and fabrication costs, and eliminates fit-up problems. Another very effective modification to this process is the conformal molding of T section spar, rib, and bulkhead webs as the faying air passage skin is cured (Figure 11). During cure, the tee cap is isolated from the skin by a Teflon separator film to

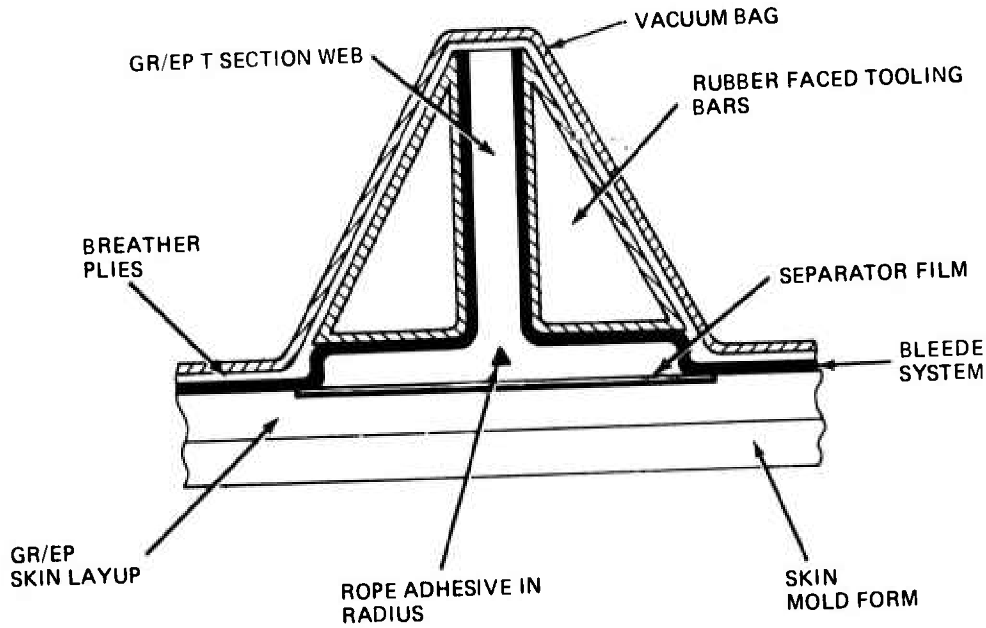


Figure 11. Conformal Molding of T Section Webs

prevent adhesion. Since the faying surfaces match, no liquid shimming is required, resulting in higher structural joint efficiency and lower assembly costs.

Another variation in the cocured laminate concept is the shell liner fuselage panels. These integrally stiffened skins are molded using segmented silicone rubber mandrels to support the hat section stiffeners during cure (Figure 12). This process has been demonstrated successfully for fuselage sections. However, a determination must still be made for each individual panel for mandrel removal considerations. Where complex contours prohibit mandrel removal, the shell liner can still be conformally cured with the skin and the panel assembled by secondary bonding. If bonding is required, manufacturing costs increase due to the extra operation and the cost of installing antipeel fasteners. Process reliability is also reduced.

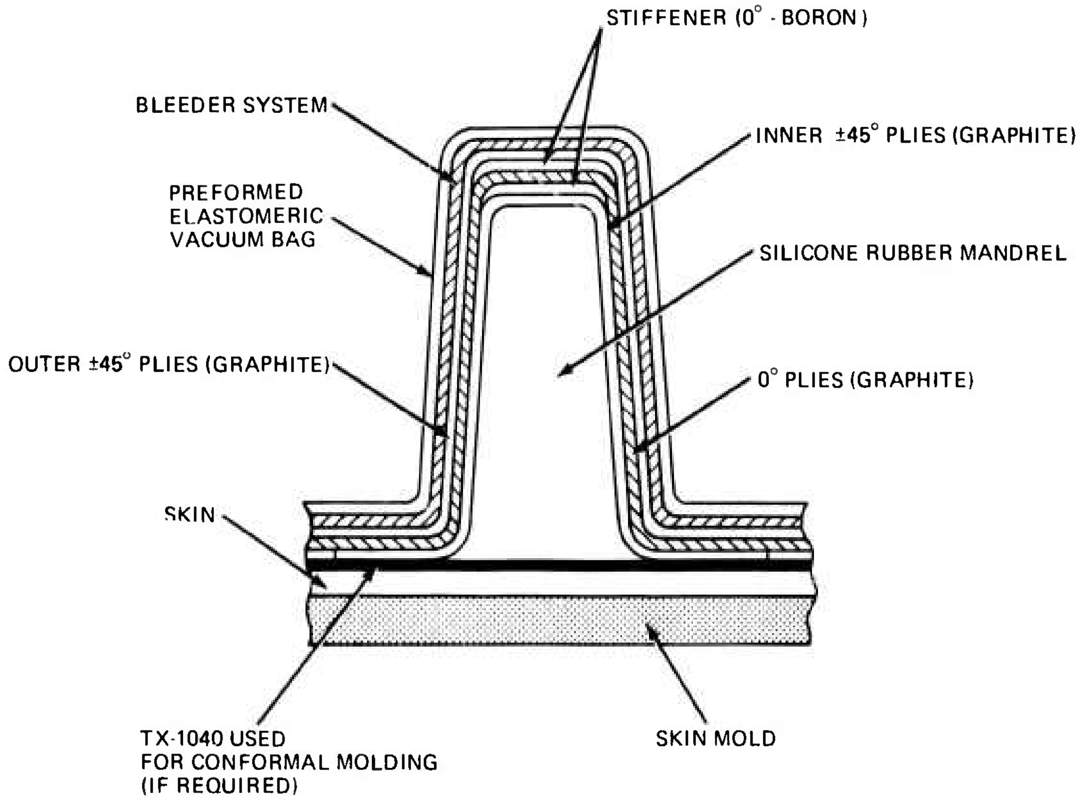


Figure 12. Shell Liner Panel Molding Concept

Thus, cocuring is the optimum manufacturing technique for solid composite skins. During this fabrication sequence, the flame-sprayed aluminum lightning protection system, metallic foil wear strips on door edges, Kevlar plies for impact resistance, and metallic fittings for lines can also be incorporated into the skin at relatively low cost. The use of these skin/attachment cocuring processes reduces the assembly fixture requirements since the location of internal parts has been fixed. Thus, the main function of the assembly fixture is to locate the various skin/attachment assemblies to the air passage contour during joining.

4.3.2 Cocured Honeycomb Panels

Honeycomb sandwich structure used in the ADCA vehicle is categorized as lightly loaded secondary aerodynamic surfaces (tips, leading and trailing edges, flaps, and rudder) and highly loaded fuselage and control surface components. Variations of the cocure process have been verified for both types of structure to eliminate prefit problems and costs inherent in secondary bonding. For lightly loaded structures, a single-step cocure process to produce the entire component is preferred. For example, the tooling for a trailing edge consists of a contoured baseplate fitted with removable blocks to form the closure webs, and a formed sheetmetal caul plate (Figure 13). Hard points are provided to

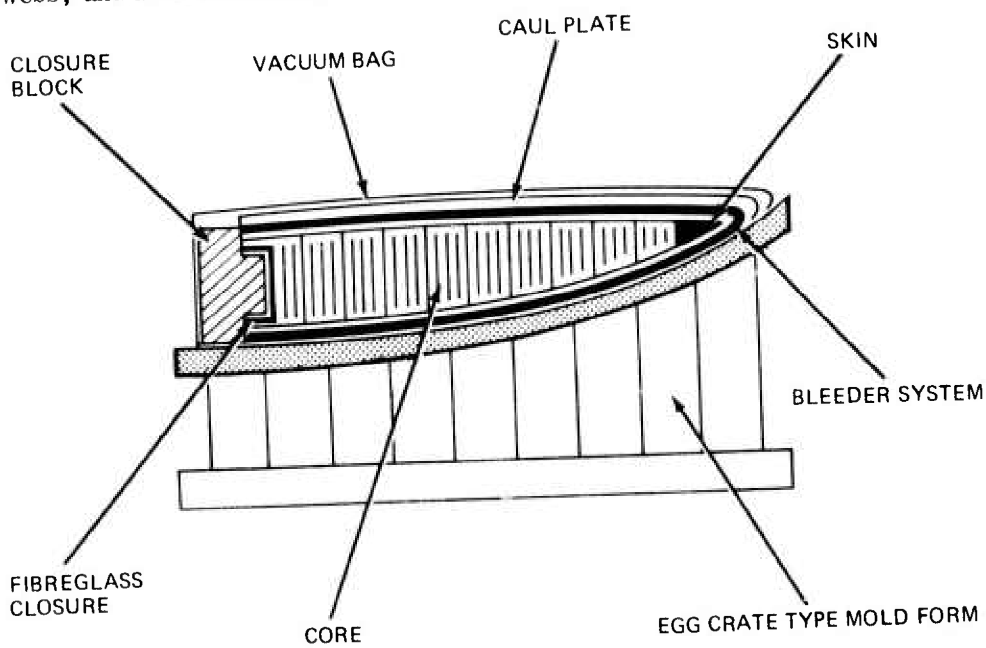


Figure 13. Layup and Tooling for Cocuring a Trailing Edge Assembly

locate the closeout fitting and closure web blocks.

As required, the skin layers are flame-sprayed for lightning strike protection prior to installation in the tool and bled through the flame spray. An alternate concept which has had mixed success involves prebleeding the skins prior to layup in the tool. A typical sequence is:

- Lay-up bleeder on tool surface, if required
- Lay-up inner skin on tool
- Apply film adhesive to inner skin
- Lay-up channel member(s) and form to shape
- Apply foam adhesive to channel sections
- Lay-up core and adhesive-coated closure member
- Lay-up film adhesive
- Lay-up outer skin on tool
- Lay-up bleeder system as needed
- Apply caul plate
- Install vacuum bag and cure
- Final trim as required

This one-step method produces an assembly from a minimum number of parts using a single tooling assembly. Controlled flow resin systems, capable of forming structural bonds, may eliminate the need for the bleeder system and film adhesive.

The one-shot cocuring process has moderate manufacturing risk, but severely restricts inspection of the completed article. As the structural requirements increase, a two-step process using one precured, fully inspected skin and a cocured skin is required. The outer or most heavily loaded skin is completely fabricated and placed in the mold. Then the core, adhesive, and inner skin are laid-up and cocured in a second autoclave cycle. This modification to the cocure process trades-off manufacturing costs for the increased inspectability required for flight-critical structures. Grumman has demonstrated the process in the fabrication of a 35-square-foot F-14 overwing fairing containing an integrally molded beam. The structure successfully passed static and fatigue tests and cost less than its metallic equivalent. Selected fuselage panels and the canards will be fabricated using this process.

4.4 SECONDARY BONDING.

Secondary bonding of composite stabilizers, rudders, spars, leading and trailing edges, and fuselage panels is a production process. The development of more uniform prepreg materials, core faying surface molds, N/C controlled core machining, and pressure pads to control the bag side surfaces of detail parts have significantly improved the control of adhesive bondlines. For highly loaded structures, secondary bonding is a preferred manufacturing process because the resultant assembly can be inspected more thoroughly. The increased structural confidence thus justifies the additional fabrication costs.

Currently available secondary bonding technology has not progressed sufficiently, however, to provide confidence that wings, large stabilizers, or large vertical fins can be structurally bonded. As a result, these structures are assembled with mechanical fasteners with penalties in weight, cost, and manufacturing flow time. In order to adhesively bond spars and ribs to a cover, a method must be developed to increase the adhesive joint's capability to transmit loads from the cover to the substructure and also redistribute the internal loads. A unique type of joint was studied to bond back-to-back channel internal spars to the lower cover in the fuel carrying center bay of the wing torque box, Figure 14. For this joint concept, dry yarns of graphite are punched through the uncured skin laminate such that two ends of yarn will result at each punch position. The number of ends per yarn and spacing of the yarns are determined as a function of the internal fuel pressure loads which the yarns and back-to-back channel section spars must resist.

The lower cover fabrication begins by applying the lightning protection system to the surface of an air passage mold. The skin plies are then laid up by the laminating center. The graphite yarns are punched through the layup by a single-needle tufting machine. Similar machines with many needles are used in the carpet industry. The fibers are held above the layup under tension by the fiber support frame and the tufts are cut to form two single yarns. The yarns are sealed with Teflon tape to prevent resin impregnation during cure.

A peel ply and the skin bleeder system are installed and the punched fibers/Teflon tape assemblies folded over the bleeder system. Contoured caul plates are then located

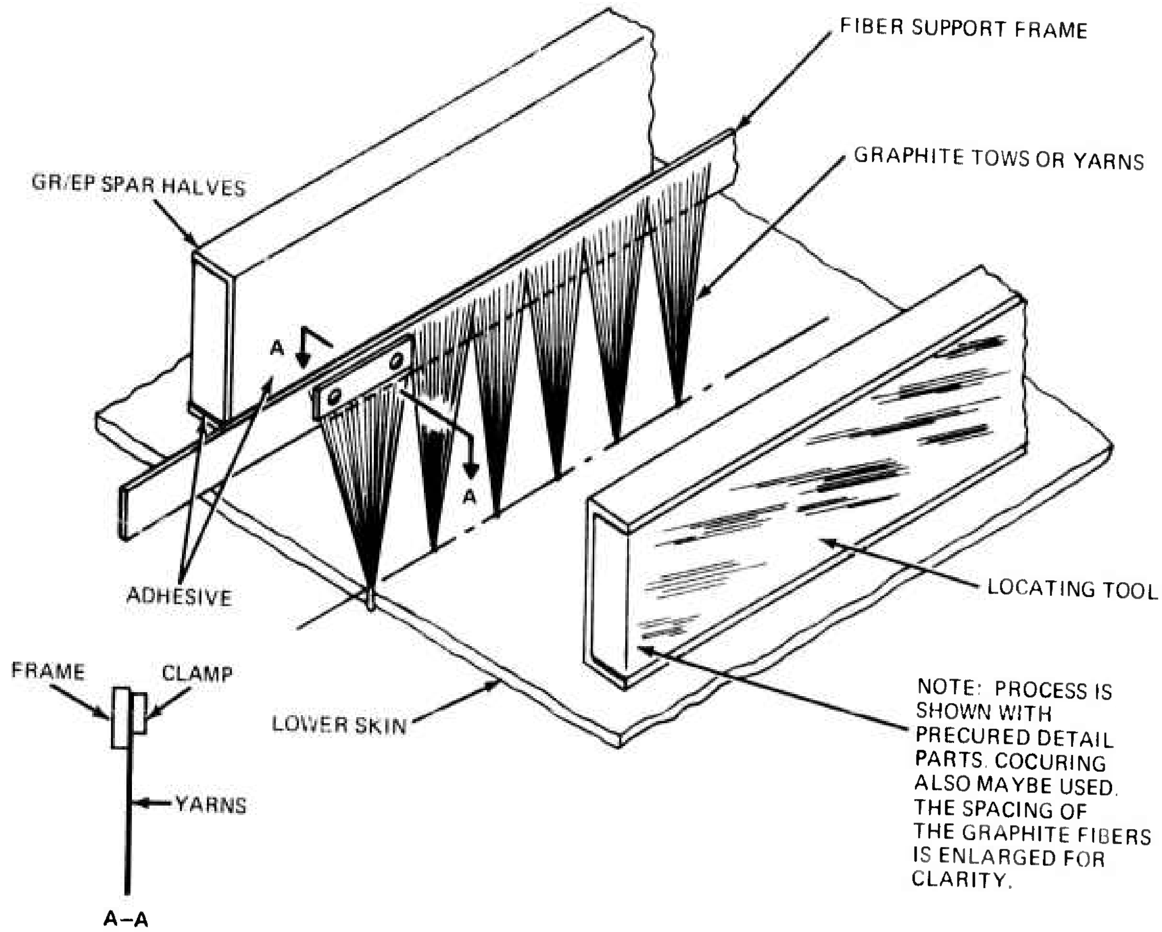


Figure 14. Lower Skin/Substructure Punched Fiber Reinforced Joint Concept

over the bleeder system to control the skin contour in areas that are subsequently bonded. Next, the skin is cured and coordination holes for subsequent operations are drilled using bushings in the mold. The skin is then inspected using standard techniques.

Concurrently, the channel sections with integral web stiffeners and reinforcements are cured using either the trapped rubber process or padded male tools fitted with WEB caul plates and elastomeric pressure pads. The tools for making these parts are mastered to the skin caul plates, and all manufacturing tolerances are forced toward the mechanically attached upper cover. The channel sections are trimmed and fully inspected.

For the assembly of the spars to the skin, film adhesive is laid up on the channel flanges and webs. The yarns molded in the skin are then combed and held upright under tension by the support frame. Next, the mating channel sections are located over the skin by hard points at the edge of the skin mold which are coordinated to the channel tools. The adhesive is cured to trap the yarns in the bonded web joint and to bond the flanges to the skin. The channel tools are removed and the assembly is inspected dimensionally and by NDT.

The baseline tooling concept has the potential problem of maintaining proper bond lines in the flange-to-cover attachment. As an alternate, the cover layup and punched fiber reinforced (PFR) assembly could be made in a low-cost plastic air passage layup tool and then transferred by the laminating center to an inner mold line steel cover mold. This concept would provide better control of the skin surface and provide a more accurate master for the substructure tooling. However, a separate air passage bonding fixture would be required.

The PFR concept is attractive because the risk in secondary bonding of precured channels is low since any voids in the bonds can easily be repaired by fastening.

Thus, the punched fiber reinforced bonded joint offers promise of significantly reducing wing assembly costs at low risk by eliminating the mechanical attachment of interior spars to the lower cover.

4.5 THERMOPLASTIC MATRIX MATERIALS

Epoxy matrices were baseline in our studies since design and process parameters are well established. However, recent feasibility studies of thermoplastic matrix advanced composites are promising and they could be an effective material for 1980 aircraft. The major disadvantage of these materials at present is the higher temperatures and pressures required for ply consolidation and forming useful shapes. However, these limitations may be overcome by utilizing appropriate technology from metalworking and automotive plastic industries.

Ideally, standard stock material of specified fiber orientation would be sheared to size and then formed to the part contour. The stock material could be made from mechanically preplied sheet consolidated by (dielectrically) heated rolls. The rolling mill is similar to that used to diffusion-bond clad metals. Shearing the sheet to size on standard equipment appears feasible. Then the parts can be formed using any of several processes. At present, die forming at elevated temperatures has been demonstrated using dies and presses usually used for titanium forming. As an alternate, room-temperature forming in metal dies has been demonstrated for the high-rate production of SAN/glass automotive parts. Possibly trapped rubber presses and high-pressure fluid pressure chambers used to form aluminum parts could be utilized with heated tools and slow rates of pressure application. Bulk forging and creep molding processes used for thick section unreinforced thermoplastic parts may also be useful. Thermoplastic parts then can be joined by various ultrasonic welding techniques which are currently available.

When more material and processing data become available for thermoplastic matrix advanced composites, a reassessment of their utilization in this program could be made.

4.6 MECHANICAL ASSEMBLY

Advanced design assembly fixtures, which precisely locate detail parts, and improved drill templates/programmed drilling machines have simplified the mechanical assembly process. The Grumman-developed five-axis drilling machine (Figure 15) has the capability of scanning the assembled structure and modifying the program to compensate for even the slightest changes in part location. Also, the ability to coordinate individual plies in a cover to the assembly fixture has been demonstrated, further reducing the possibility of misplaced holes. Carbide drill/countersink cutters, used with a variety of drilling machines operated at precise speeds and feeds, have demonstrated the capability of drilling and countersinking Gr/Ep cover-to-substructure attachment holes in one operation without breakout. Ultrasonic drilling of boron/epoxy laminates with refurbishable plated diamond grit tools also has been demonstrated to be a low-cost production process. Lastly, the development of readily processible liquid shim materials and blind bolts which develop large footprints have further simplified mechanical assembly.

4.7 SUMMARY

In summary, a broadbased manufacturing technology exists to ensure efficient ADCA vehicle fabrication. Recent industry wide experience with large-scale structural components has demonstrated the structural reliability and cost effectiveness of this technology. Also, the concept of the design-manufacturing interface to improve the producibility of composite structures has been demonstrated on various programs. The use of the established processes, coupled with the advanced processing approaches (translaminar reinforcement, conformal molding, and punched fiber reinforced joints) make possible the production of Advanced Design Composite Aircraft in the near term at affordable cost.

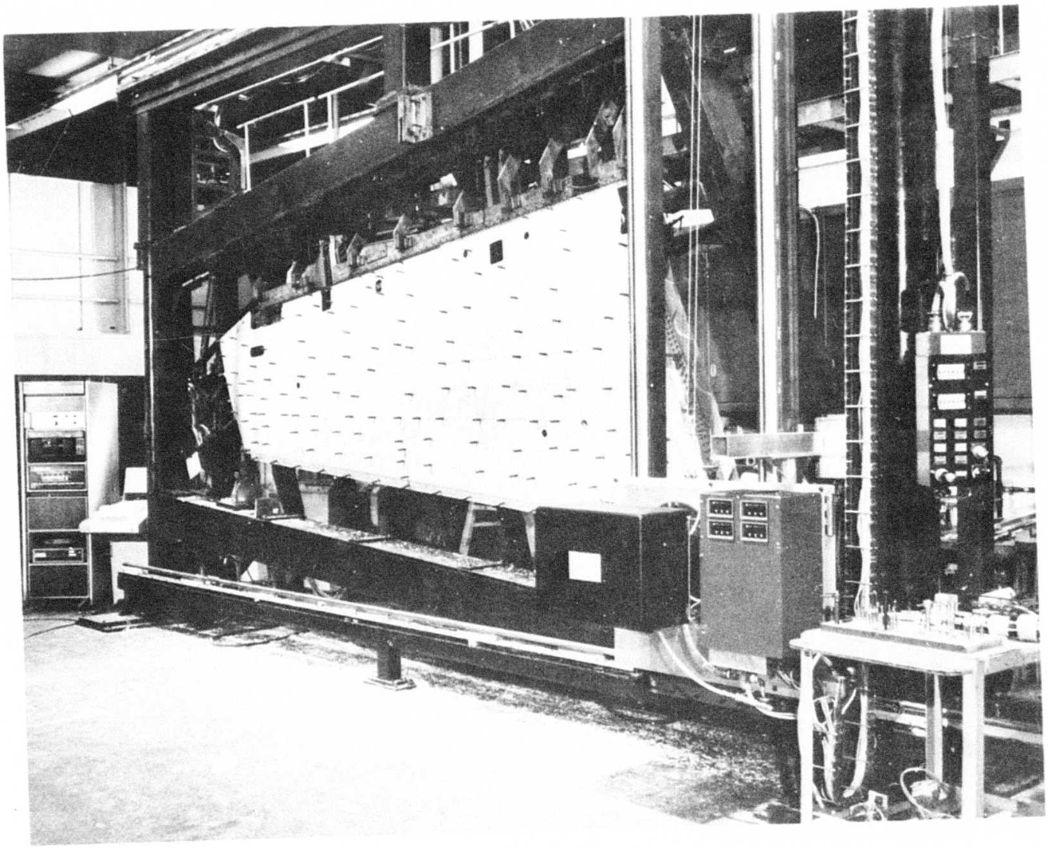


Figure 15. Five-Axis Drilling Fixture

Section V

SUBASSEMBLY MANUFACTURING PLANS

The vehicle was divided into four major subassemblies: wing, vertical fin and rudder, canard, and fuselage. Alternate manufacturing concepts were evaluated for each subassembly to reduce risk and cost. The following manufacturing plans were then derived.

5.1 WING FABRICATION

The wing box assembly is configured as a single-unit multi-spar structure with a fuel tank in the center bay, Figure 16. The entire wing is fabricated from composite materials except for titanium ribs at BL 120.95 and BL 196.60. Also, six titanium fittings are provided for attachment to the fuselage, Figure 17. The one-piece covers extend from tip to tip and vary in thickness from 44 to 129 plies. The orientations have been selected to yield both maximum structural efficiency and increased wing twist for improved vehicle performance. The majority of the connections to the lower cover are made using either adhesive bonding or integral molding, with only the front, rear, and mid main spars and ribs being mechanically attached. The wing torque box is closed out by fastening the upper cover to the substructure. Titanium Hi-locks with A-286 CRES steel collars are used in the open joints, with CRES or titanium Jo-bolts used to attach the blind upper cover. All fasteners are installed with wet sealant to prevent corrosion. O-ring seals under the fastener heads prevent fuel leakage. The fuel tank is sealed with fiberglass/epoxy plates and sealing grooves. Sealing the tank by fusing thermoplastic sheets bonded to both covers and substructure caps were also considered and should be evaluated further.

Three concepts of attaching the interspars to the lower cover were studied. These are:

- Punched Fiber Reinforced Joint concept shown in section A-A of Figure 16 (sheet 2)
- Integrally molded caps with mechanically attached T-section webs shown in alternate section A-A of Figure. 16 (sheet 2)
- Advanced Wing Box Concept, Figure 18

These various design concepts utilize advanced cocuring/bonding concepts to lower fabrication costs by significantly reducing part/fastener count compared to current mechanically fastened structures. The fabrication of the wing box using these design concepts is discussed in the following subsections.

5.1.1 Fabrication of Wing Common Detail Parts

Various detail parts are used in all three wing designs; their fabrication is discussed below.

5.1.1.1 Front, Rear, and Mid Spar Fabrication

These highly loaded assemblies are flight-critical. In order to enhance the inspectability and reduce manufacturing risk, these components are fabricated by assembly bonding of detail parts. The channel sections are cured on male steel molds provided with caul plates in the web areas and pins to provide coordination. The cap strips are molded on contoured plates. The open angle channel mold is fitted with side plates and used as a bonding fixture. The detail parts are pinned to provide coordination for bonding. The adhesive bond line thickness is measured by the adhesive isolation technique to determine the adhesive layup.

5.1.1.2 Composite Rib Fabrication

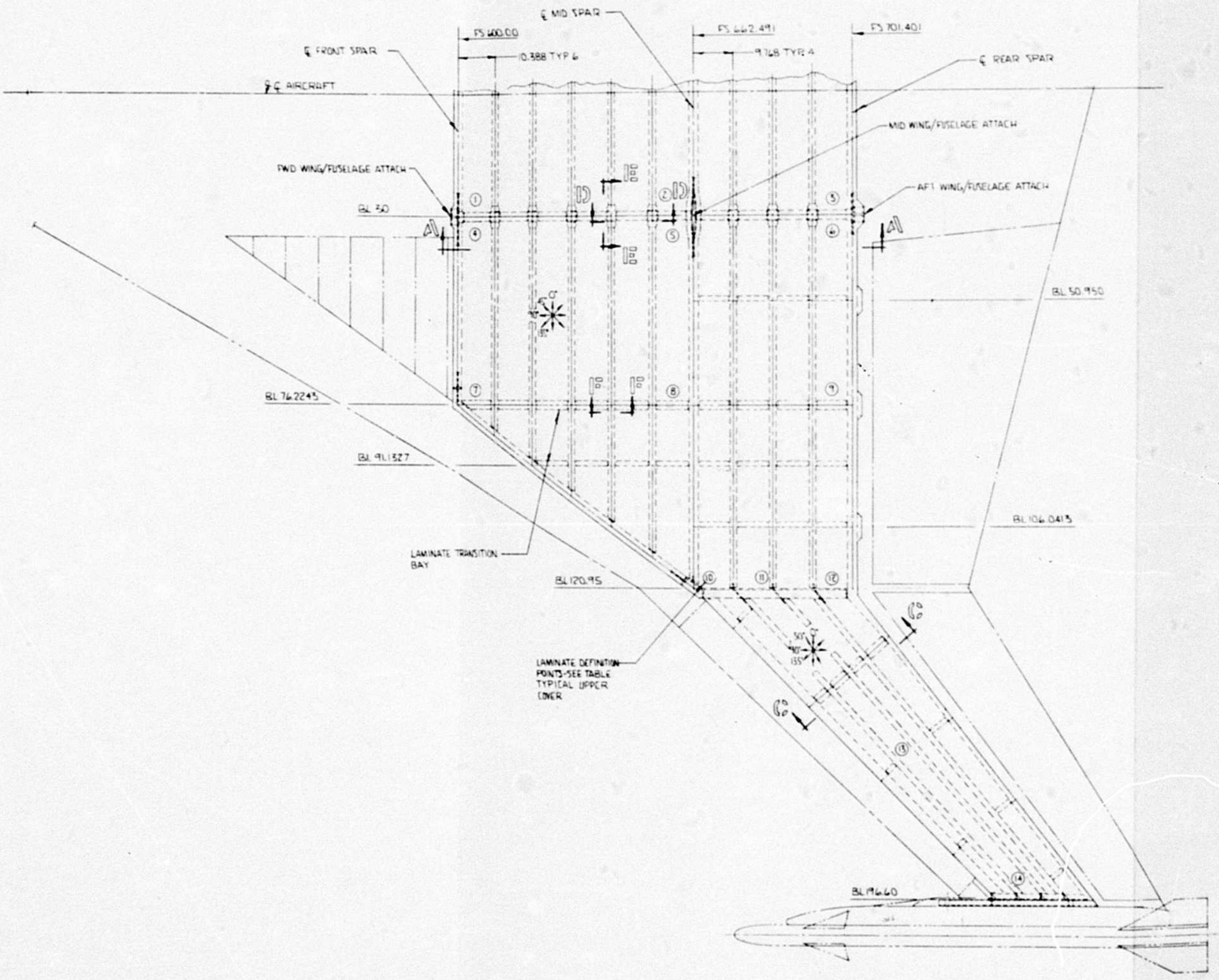
The composite ribs are molded on mated rubber-faced metal tools. The channel section bleeder system and layups are placed on their respective mold halves and the flanges are formed under vacuum. The layup/mold assemblies are joined, and the adhesive required to fill the radius is installed. Then the cap strip layups and bleeder system is installed. The cap caul plates are added, and the assembly is envelope-bagged prior to autoclave cure and oven post-cure.

5.1.1.3 Fabrication of Composite Clips

The Gr/Ep clips required are machined from cross-ply pultrusions. As an alternate, the clip layups are cut from preplied sheet and autoclave-cured on male tools sized to produce multiple parts.

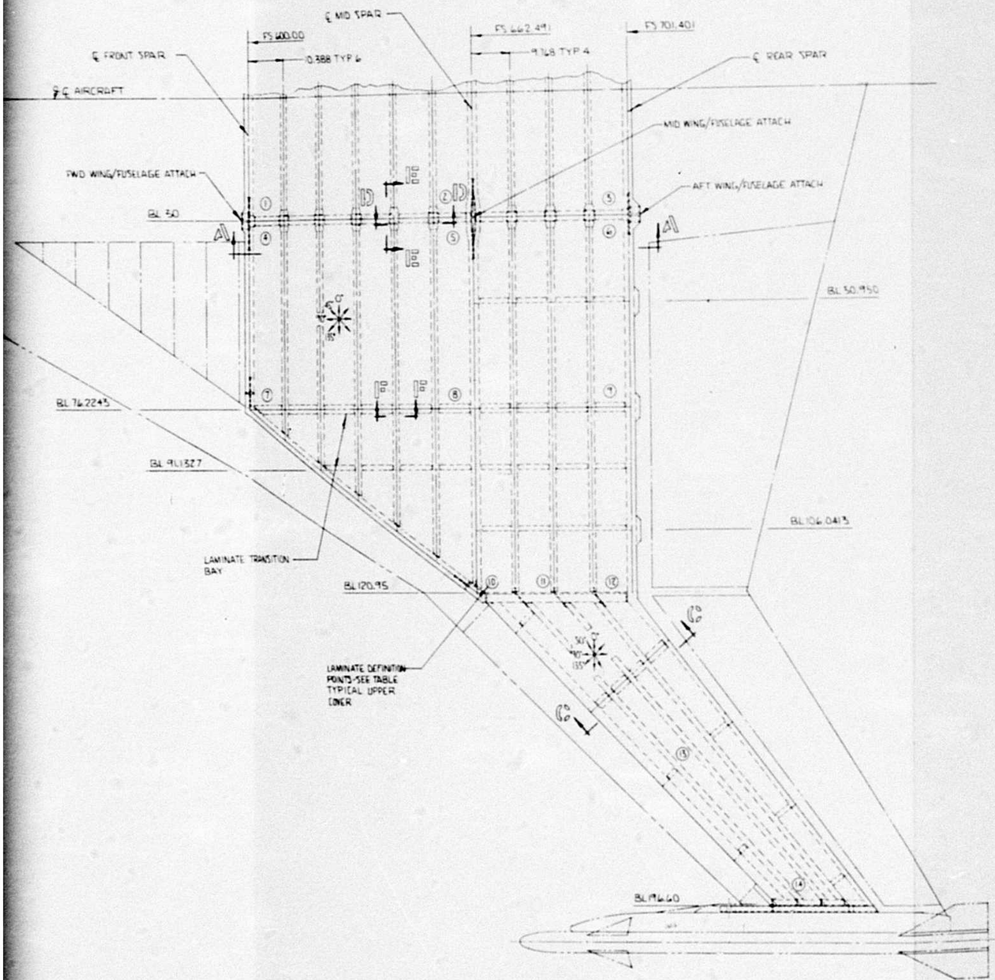
5.1.1.4 Fabrication of Titanium Detail Parts

The six wing-to-fuselage attachment fittings, the ribs at BL 120, and two tip ribs are machined from ultrasonically inspected, fracture-controlled plate stock. These parts are primed and painted with linear polyurethane paint prior to installation.



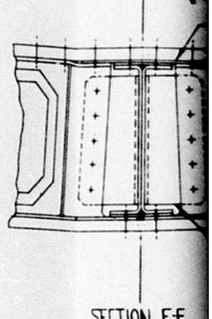
POINT	NUMBER OF PLIES	
	0° - 60° - 45° - 90° - 135°	TOTAL
1	29 — 8 - 5 - 8	50
2	46 — 10 - 5 - 10	71
3	43 — 8 - 5 - 8	64
4	22 — 11 - 6 - 11	50
5	58 — 21 - 6 - 21	86
6	41 — 14 - 6 - 14	75
7	12 — 15 - 6 - 15	48
8	30 — 25 - 6 - 25	86
9	35 — 18 - 6 - 18	77
10	30 - 60 — 19 - 6	115
11	28 - 76 — 19 - 6	12[illegible]
12	28 - 50 — 19 - 6	103
13	23 - 30 — 23 - 4	80
14	20 - 2 — 20 - 2	44

Figures 16. ADCA Win[illegible]
(Sheet 1 of [illegible]



POINT	NUMBER OF PLIES 0° - 60° - 45° - 90° - 135°	TOTAL
1	29 — 8 - 5 - 8	50
2	46 — 10 - 5 - 10	71
3	43 — 8 - 5 - 8	64
4	22 — 11 - 6 - 11	50
5	38 — 21 - 6 - 21	86
6	41 — 14 - 6 - 14	75
7	12 — 15 - 6 - 15	48
8	30 — 25 - 6 - 25	86
9	35 — 18 - 6 - 18	77
10	30 - 60 — 19 - 6	115
11	28 - 76 — 19 - 6	129
12	28 - 50 — 19 - 6	103
13	23 - 50 — 23 - 4	80
14	20 - 2 — 20 - 2	44

Figures 16. ADCA Wing Structural Concept (Sheet 1 of 2)



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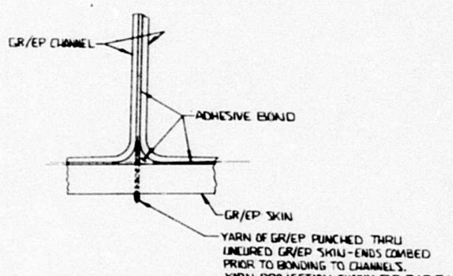

DETAIL-B
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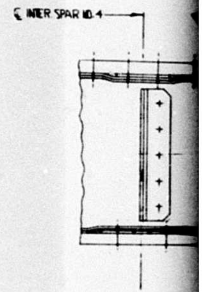
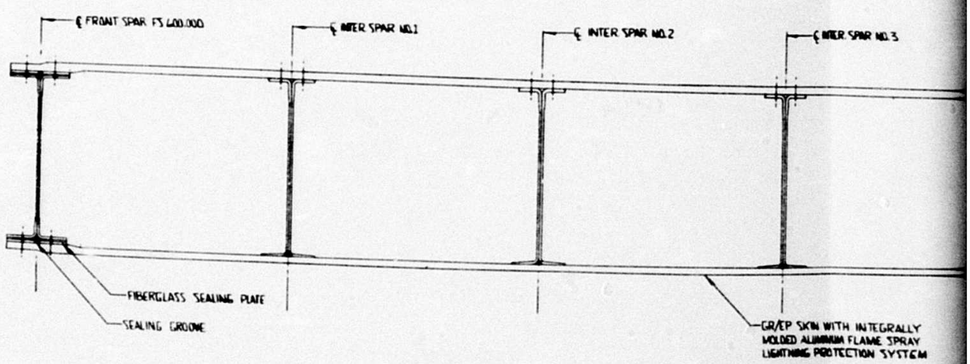
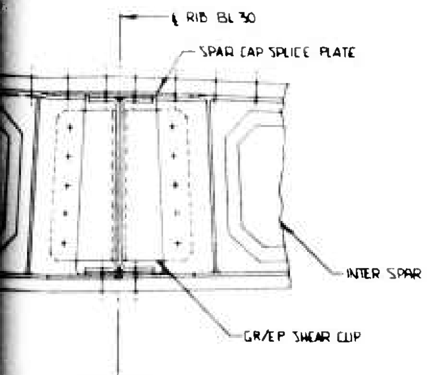
SCHEMATIC







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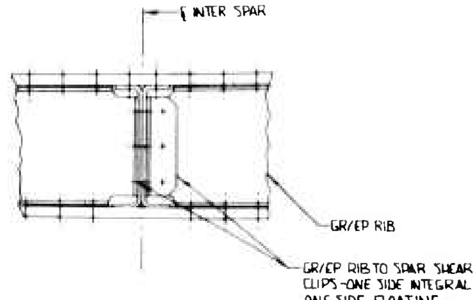

SECTION F-F
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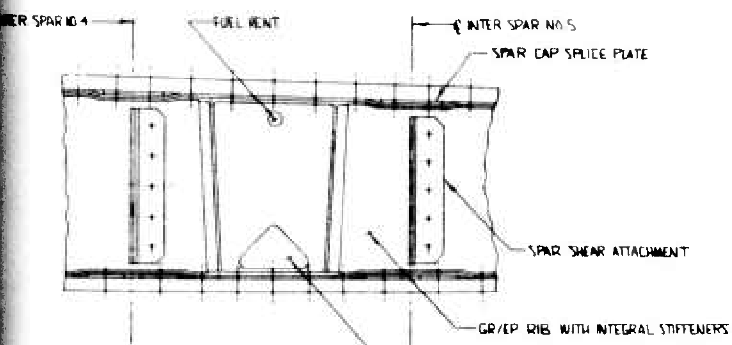

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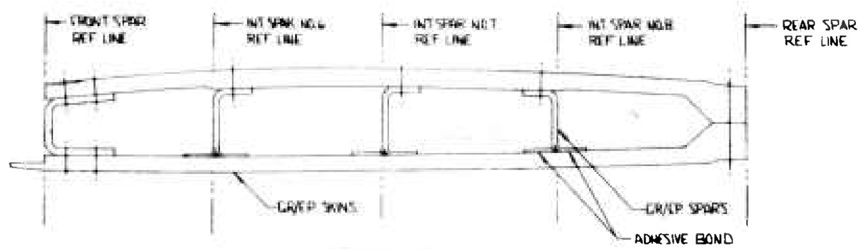

SECTION C-C
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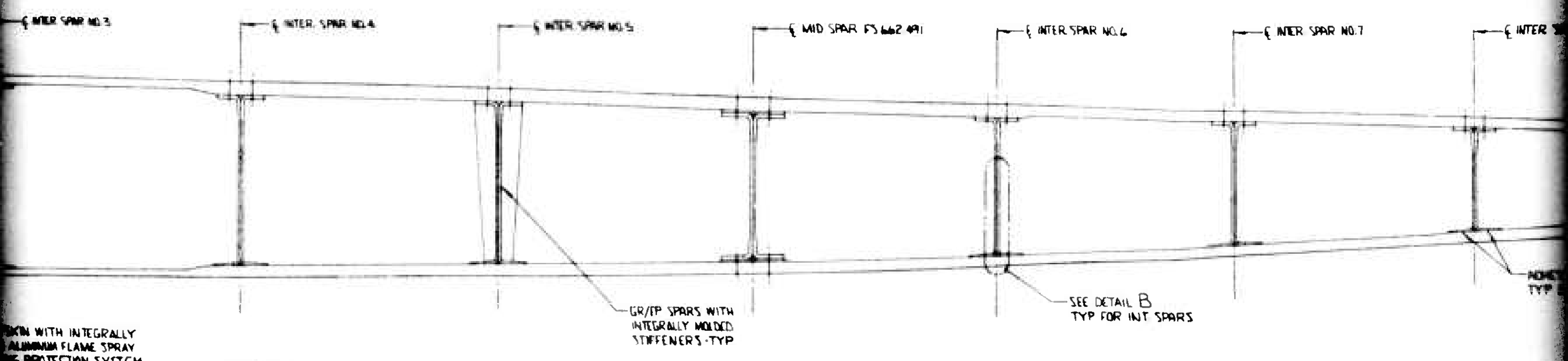

SECTION A-A
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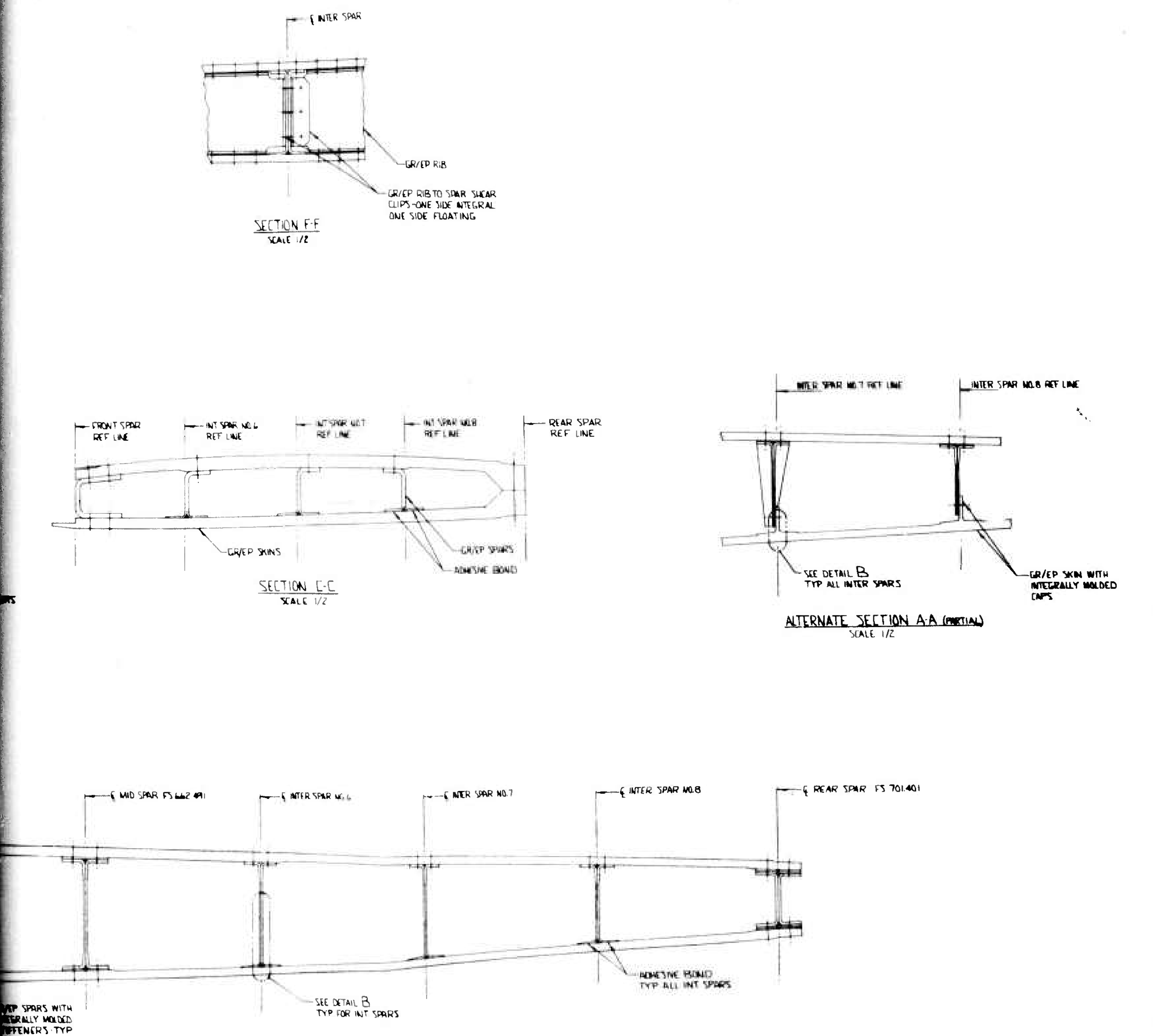


Figure 16. ADCA Wing Structural Concept (Sheet 2 of 2)

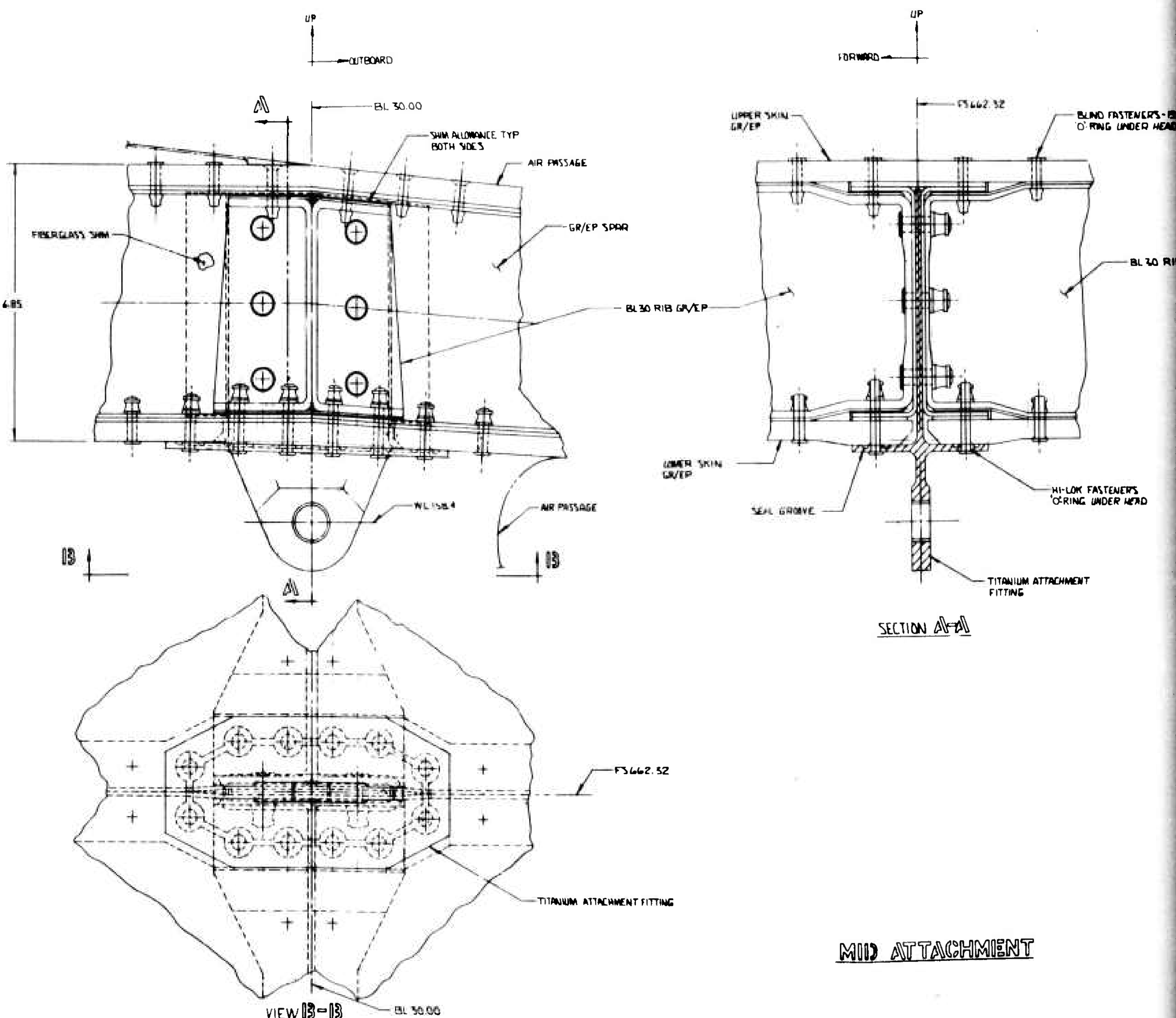


Figure 17. ADCA Wing/Fusel
(Sheet 1 of 3)

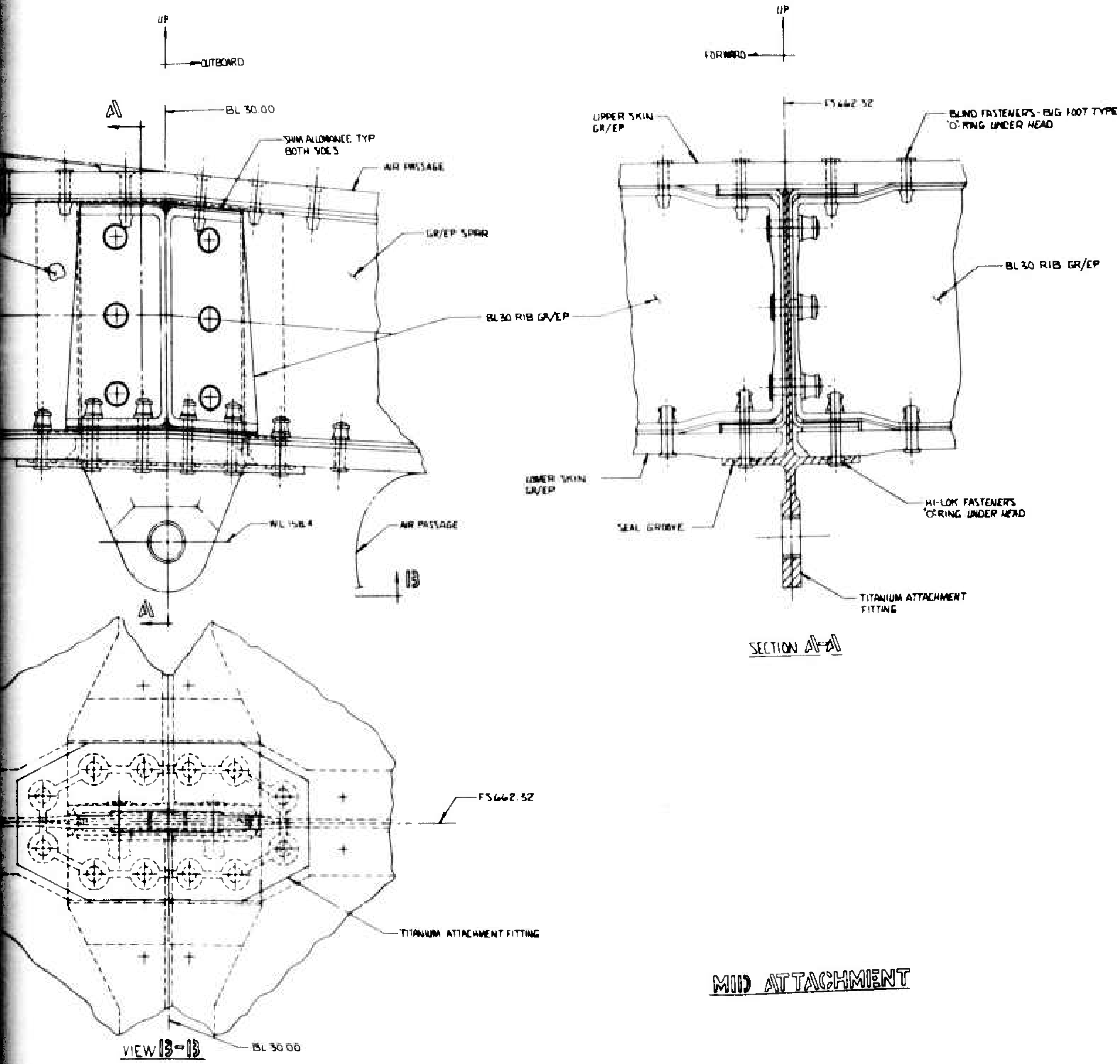
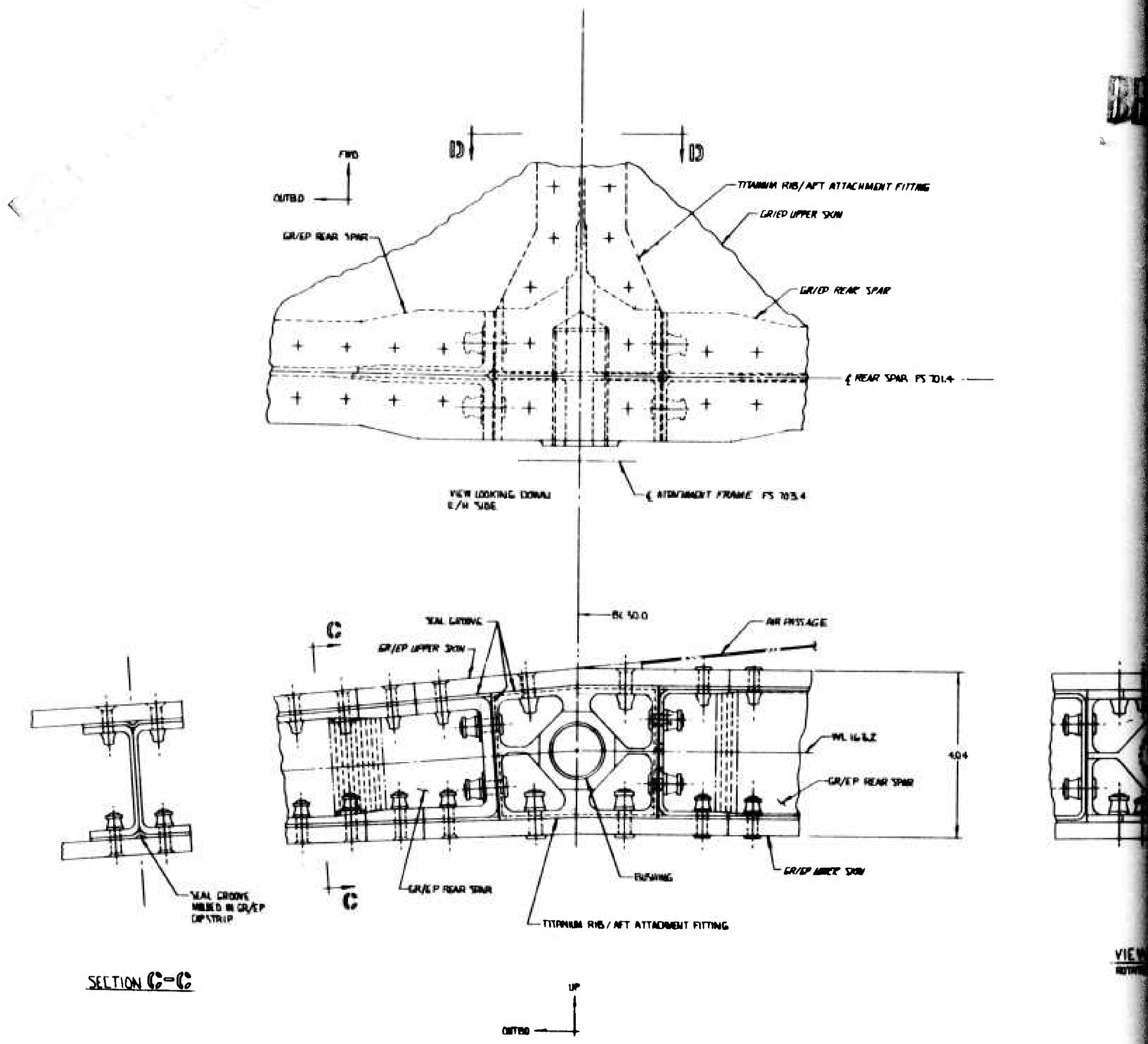
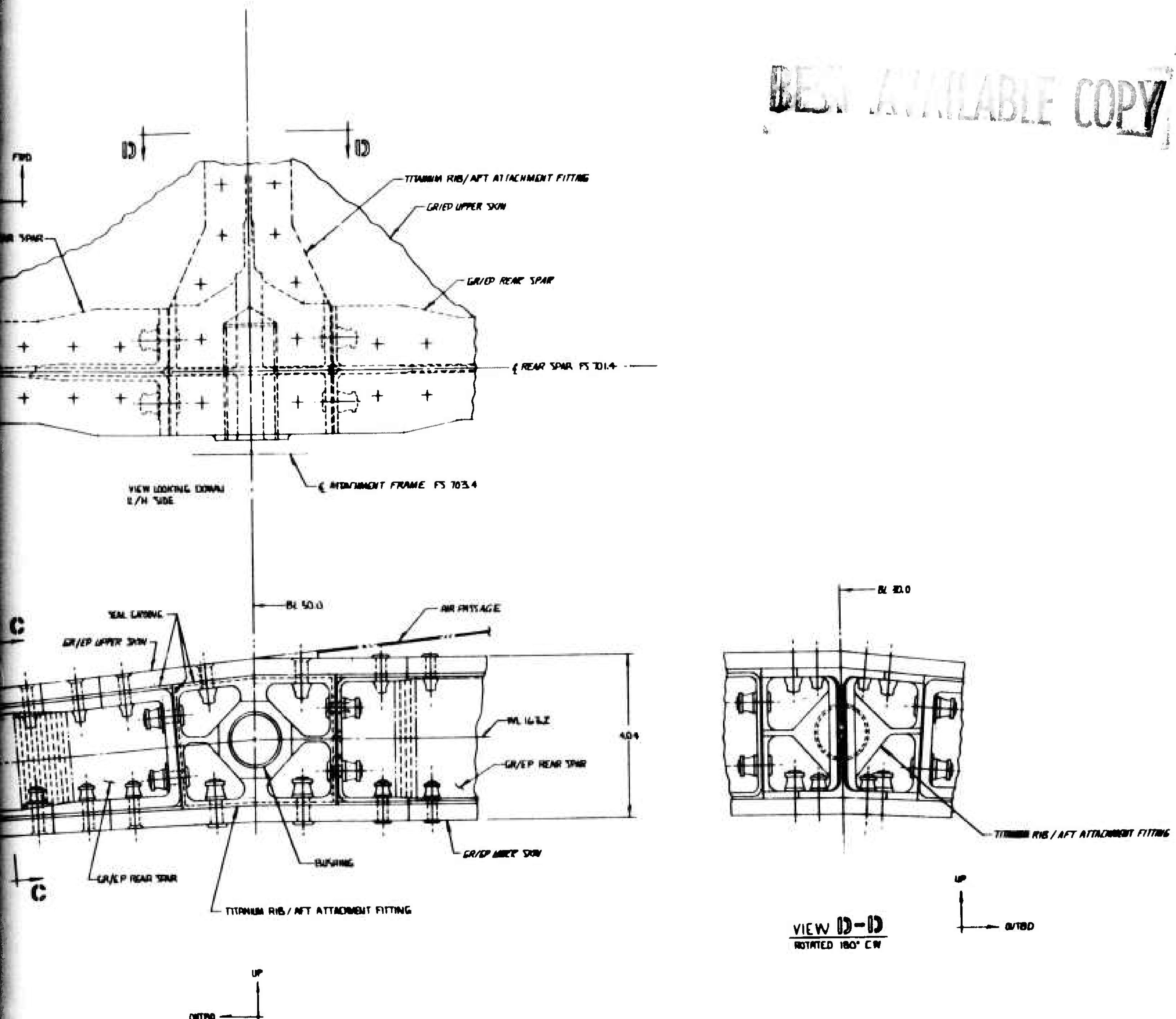


Figure 17. ADCA Wing/Fuselage Attachment
(Sheet 1 of 3)



AFT ATTACHMENT

Figure 1



AFT ATTACHMENT

Figure 17. ADCA Wing/Fuselage Attachment
(Sheet 2 of 3)

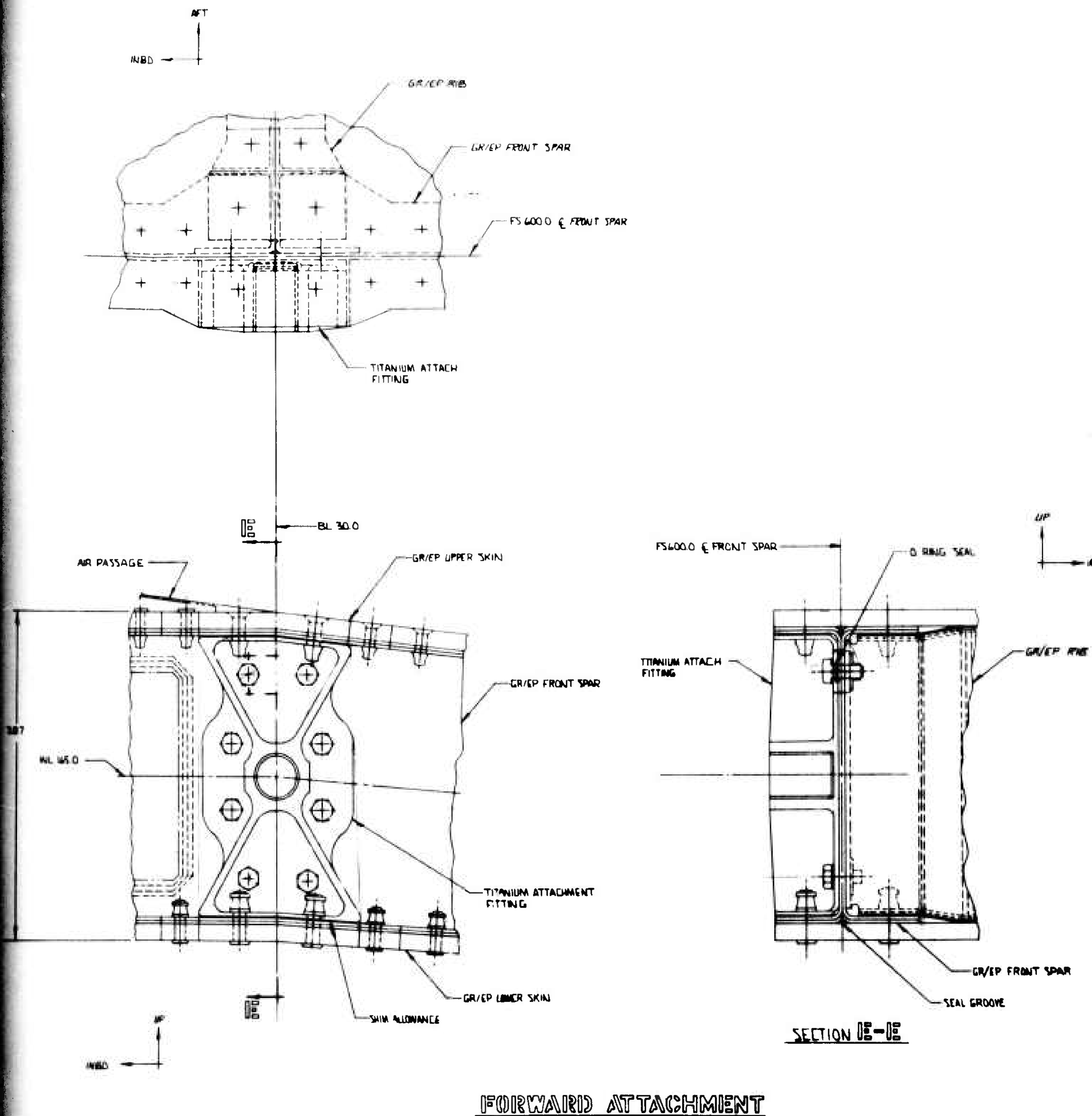


Figure 17. ADCA Wing/Fuselage Attachment
(Sheet 3 of 3)

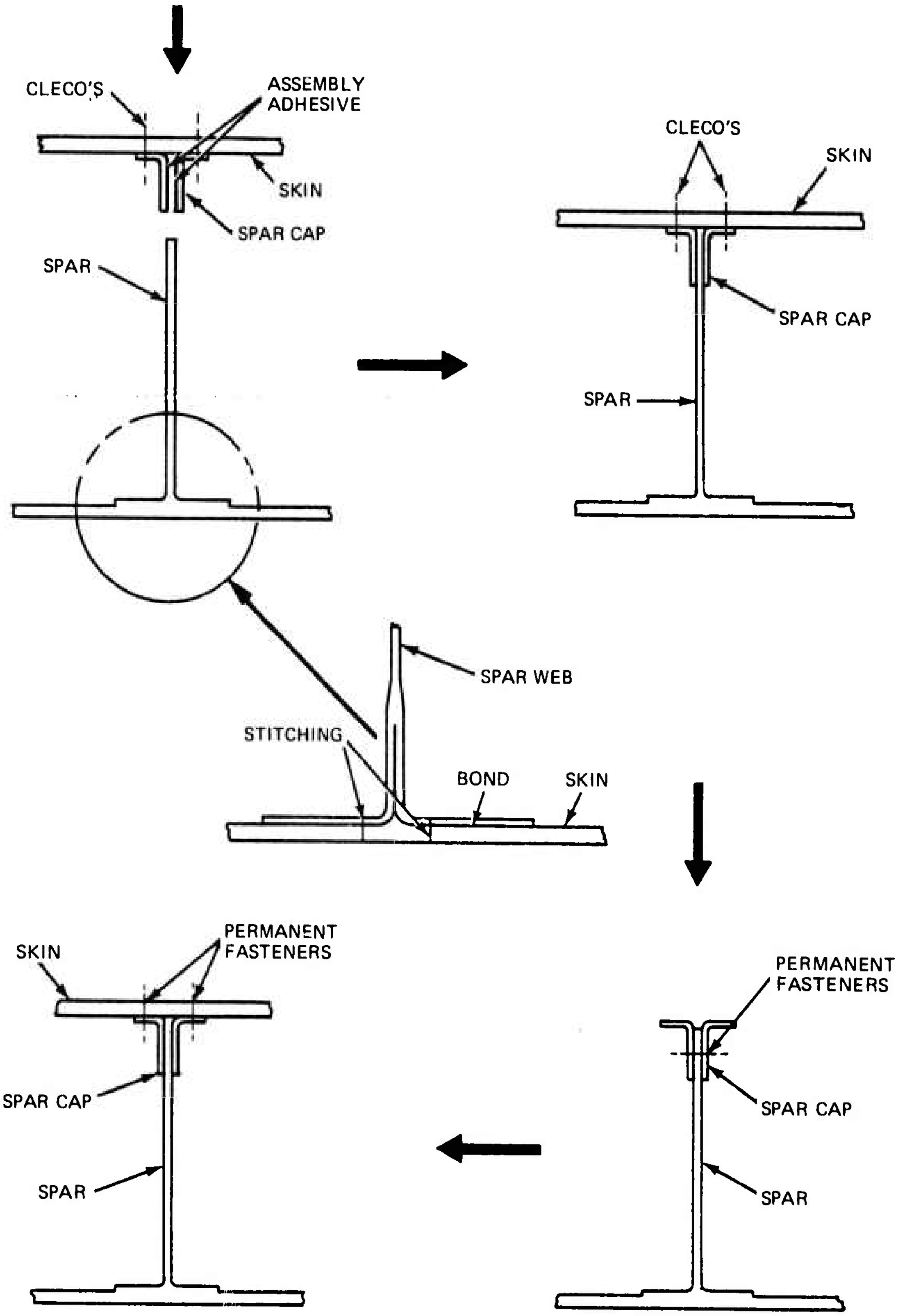


Figure 18. Advanced Wing Box Concept

5.1.1.5 Leading Edge, Trailing Edge, Tip Cap, and Wing Flaperon Fabrication

These structures all are fabricated using the single-step cocure process described in Subsection 4.3.2. Metallic surface protection systems for lightning strike or impact resistance will be cocured to the skins as needed.

5.1.2 Fabrication of Punched Fiber Reinforced Joint Design Wing Box

The punched fiber reinforced joint design, section A-A of Figure 16 (sheet 2), has the advantage of producing an assembly of the lower cover and interior spars with a single major tool. The process sequence is described in Subsection 4.4. The front, rear, and main mid spars and ribs are attached to the interior spars by clips and fastened to the lower cover to complete the substructure/ lower cover assembly. The upper cover is then attached by blind fasteners. The five-axis drilling fixture, Figure 15, is used to drill and countersink the holes for all cover-to-substructure attachments.

The substructure buildup may be done in two ways depending on production rates. On a prototype basis, the hard points at the edge of the skin mold are used to coordinate installation fixtures which control the locations of the remaining spars, ribs, and clips. After checking the fit to the lower cover, pilot holes are drilled for the spar-to-rib attachments and undersize temporary hardware installed. The assembly is then placed in the five-axis drilling fixture and scanned to generate the tape which controls the hole pattern in the lower cover. The lower cover-to-substructure attachment holes are drilled/countersunk in one operation. The attachments are liquid-shimmed, permanently fastened, and sealed as needed. The assembly is then removed from the five-axis fixture, and the spar to rib attachments are completed as a bench assembly. The wing attach fittings, Figure 17, and internal systems are installed. The assembly is then returned to the five-axis fixture, and the upper cover to substructure attachment holes are drilled and countersunk. After liquid shimming and sealing, the upper cover is attached by titanium Jo-bolts. The leading edge, trailing edge, and tip cap are installed to complete the assembly process.

For higher production rates, a separate bonding fixture is required to bond the interspars to the lower cover. Also, the five-axis drilling fixture is upgraded to provide for locating the nonbonded spars and ribs. Lastly, installation fixtures coordinated from the front and rear spars are used for drilling the spar to rib attachments.

5.1.3 Fabrication of Wing Box with Integral Caps and T-Section Spar Webs

This design, shown in alternate section A-A of Figure 16 (sheet 2), is identical in basic concept to the punched fiber reinforced joint except that the spar cap at the lower skin is integrally molded with the skin. This concept has the advantages of air passage tooling, relaxed tolerances on spar height control for subsequent assembly operations, and elimination of the secondary bond cycle. The one disadvantage is that an extra line of fasteners is required at each interspar to attach the T-section web to the integrally molded upstanding cap leg.

The lower cover mold form is of steel eggcrate design except that the edges are provided with hard points to control the bar tooling required to form the spar caps, Figure 11. The cover layup sequence begins by preparing the flame spray lightning protection system. Next, the basic cover skin is laid up by the laminating center. Concurrently, the angles comprising the cap are laid up by blanking strips from multiple-ply prelaid sheet. The punched yarns are installed. The angle bleeder system is installed on the bar tools and the composite layup is formed. The mating bar tools are located over the cover layup so that the punched yarns are in the bondline. The cover bleeder system is then installed. The assembly is bagged with a preformed silicone rubber bag and autoclave-cured. Tooling holes, located by bushings in the mold, are drilled in the cover prior to post-cure and provide coordination to the assembly fixture. The angles are then trimmed as required before the assembly is inspected.

The upper cover and T-section webs are molded concurrently using the conformal molding process described in Subsection 4.3.1. Coordination holes for the cover-to-web attachments are drilled prior to removal from the mold for post-cure. The parts are trimmed as required prior to inspection.

The assembly fixture, consists of two major parts. The lower cover frame supports the lower cover assembly on contour boards in the location controlled by the tooling holes and provides hard points for locating the I beam spars and tip ribs. The upper cover frame has two sets of contour boards to which either the upper cover or T-web assemblies can be attached. The space between the upper and lower frames is maintained by precision blocks to establish the chord height. Shot bags are used to push the lower cover against the contour boards since the lower cover is blind at start of the assembly process. The assembly fixture is set up to locate the T-section webs relative to the integrally molded angles. Then the remaining spars and ribs are loaded into the assembly fixture and prefitted. The location of the T-webs is corrected as required, and permanent attachment to the caps is made.

Then the front, mid, and rear spars and ribs are temporarily attached to the lower cover. The attachment holes are drilled through the lower cover by the five-axis machine, and the joints are shimmed and permanently fastened. The spar-to-rib connections are then made, and the fuselage attachment fittings and systems installed.

The assembly is returned to the five-axis drilling fixture and the upper cover/substructure attachments are drilled. After liquid shimming, the upper cover is permanently installed. Attachment of the leading and trailing edges, tip cap, and flaperons complete the assembly process.

5.1.4 Advanced Wing Box Concept Fabrication

The advanced wing box concept, Figure 18, consists of an integrally molded lower cover, spar cap and spar web, upper spar cap, and upper cover. The lower cover/spar cap attachment is reinforced by a continuous chain stitch prior to cure, as described in Subsection 4.3.1. The upper cover and upper spar cap are cured using the conformal molding process. This concept has the advantage of simplifying the spar-to-rib coordination before the lower cover is drilled, and uses a less complex assembly fixture. An advanced wing box assembly technique is also utilized with the concept, in an effort to further reduce assembly cost by minimizing any final upper skin to spar shimming operation. The technique makes use of a quick curing "assembly adhesive", temporary assembly hardware (Cleco's), and the assembly fixture shown in Figure 19. The sequence of operations would be as follows:

- Locate upper spar caps to upper skin and pilot drill
- Attach caps to skin with Cleco's
- Apply assembly adhesive to caps
- Locate upper skin cap assembly to previously cured lower skin/integral web assembly
- Remove Cleco's and upper skin
- Install permanent cap to web fasteners and make spar-to-rib joints
- Replace upper skin and install permanent skin to cap blind fasteners

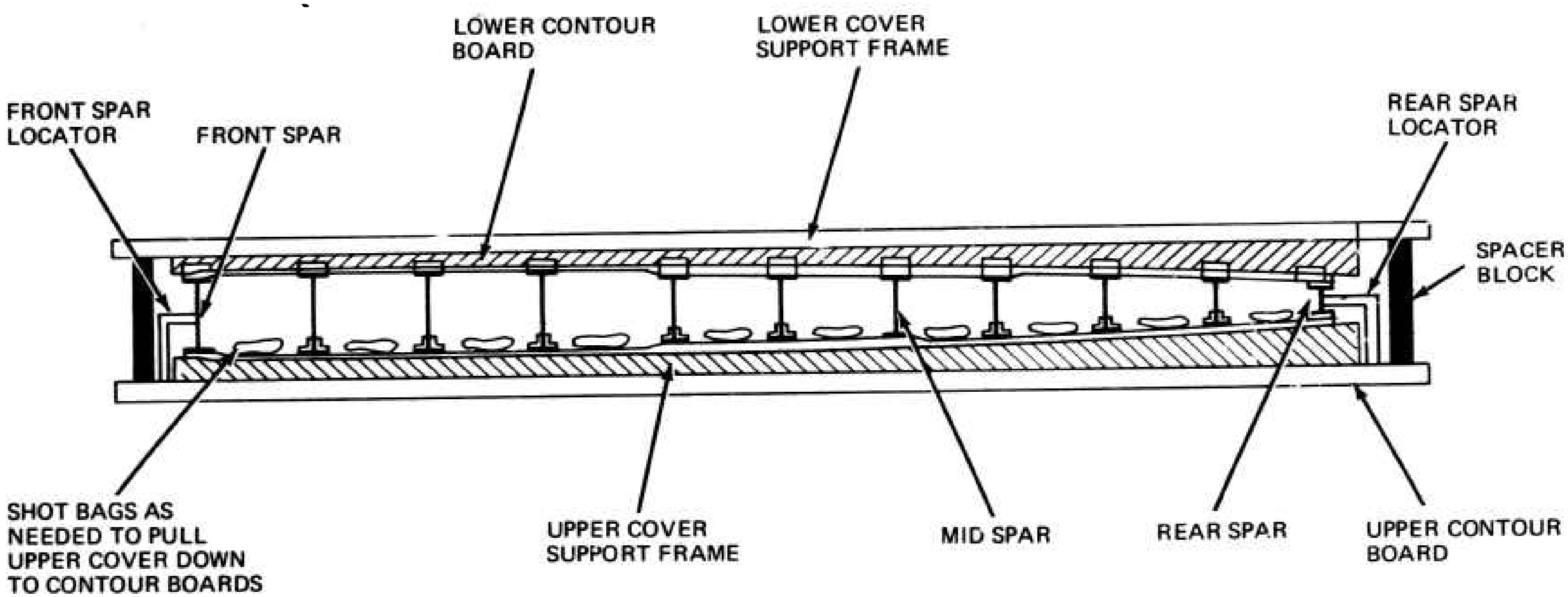


Figure 19. Wing Assembly Fixture

5.1.5 Summary

The three design concepts discussed above all make use of advanced manufacturing technology to reduce part and fastener count, resulting in cost savings. Although detailed cost estimates were beyond the scope of this study, all three concepts will cost less to manufacture than current mechanically fastened structure. The critical problems to be addressed are the tooling/design interaction to reproducibly produce high-strength integrally molded spar webs, and how to extend these concepts to include the ribs. Several current programs are addressing the tooling problems, and the data obtained would serve as a springboard for ADCA production.

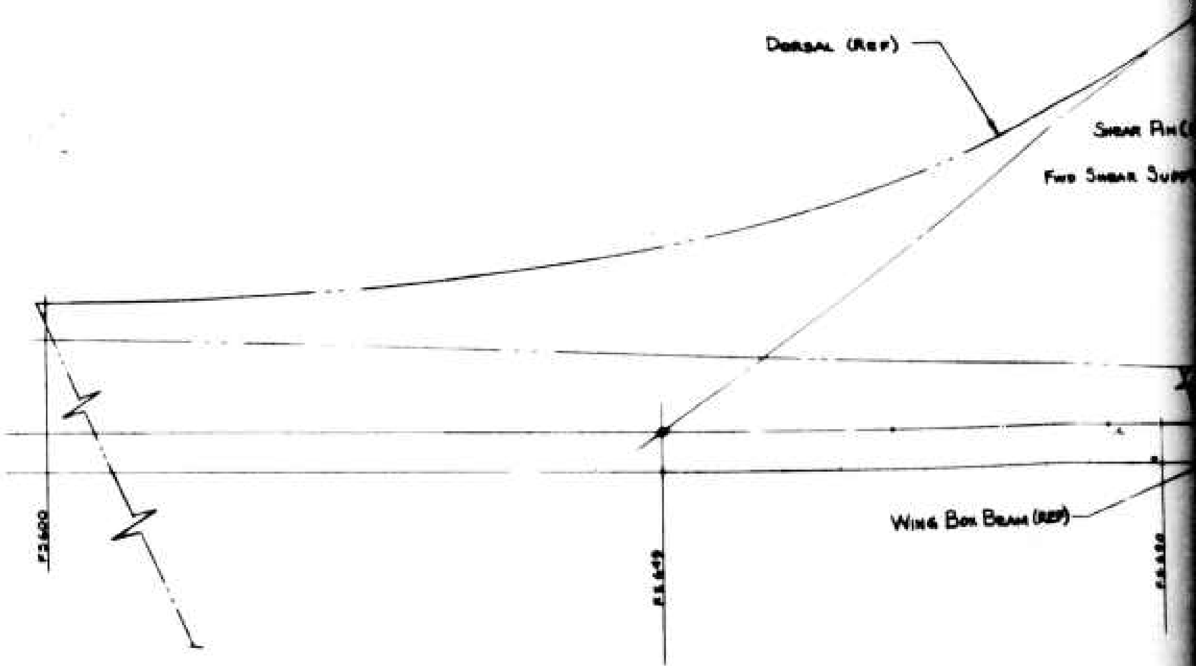
5.2 VERTICAL AND DORSAL FINS AND RUDDER FABRICATION

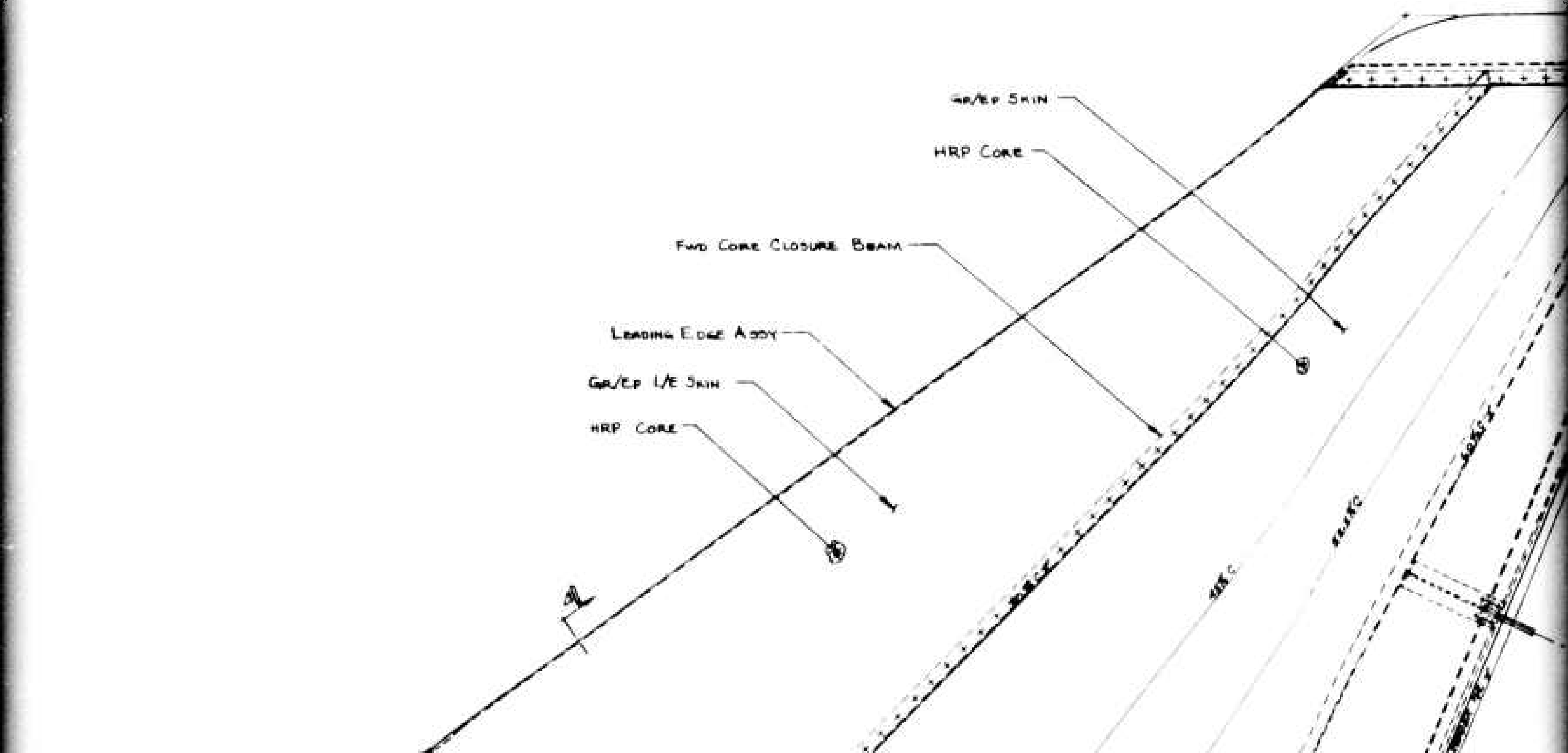
The vertical fin assembly consists of a structural torque box, leading edge assembly, tip cap, and provisions for joining to the fuselage, rudder, and dorsal fin (Figure 20). The torque box consists of Gr/Ep spars, ribs, and covers, HRP core, and various attachment fittings. Since the torque box contains a structural mid spar, the assembly is not applicable to the one-shot cocuring process. As an alternate, cocuring the skins over a prefabricated substructure was considered and discarded due to high manufacturing risk and less inspectability than desired for a highly loaded, flight-critical assembly. Therefore, the torque box will be assembly bonded from precured details using procedures verified in the fabrication of 250 ship sets of F-14 stabilizers. The leading edge and tip cap will be cocured in one operation. The dorsal fin will also be cocured in one operation. The rudder, however, will have a fully inspected, precured front spar joined to the skins and core in a second bonding cycle.

5.2.1 Fabrication of the Vertical Fin Torque Box

The cover skins are laid up by the laminating center on sculptured male core faying molds and flame-sprayed prior to cure. The intercostal rib and spars are net-molded on male tools and the hinge support riblets are molded using mated dies.

The spars and ribs are prefitted and pilot-drilled; they are then packed with core as required. Next, the over-thickness core details are prefitted, and the assembly is bonded-with foam adhesive in a fixture under mechanical pressure in an oven. The hinge support riblets, hinge fittings, and aft closure beam are then permanently installed.


DORSAL (REF)
WING BOX BEAM (REF)



FIN (ASSY & INSTL)
1/4 SCALE

2

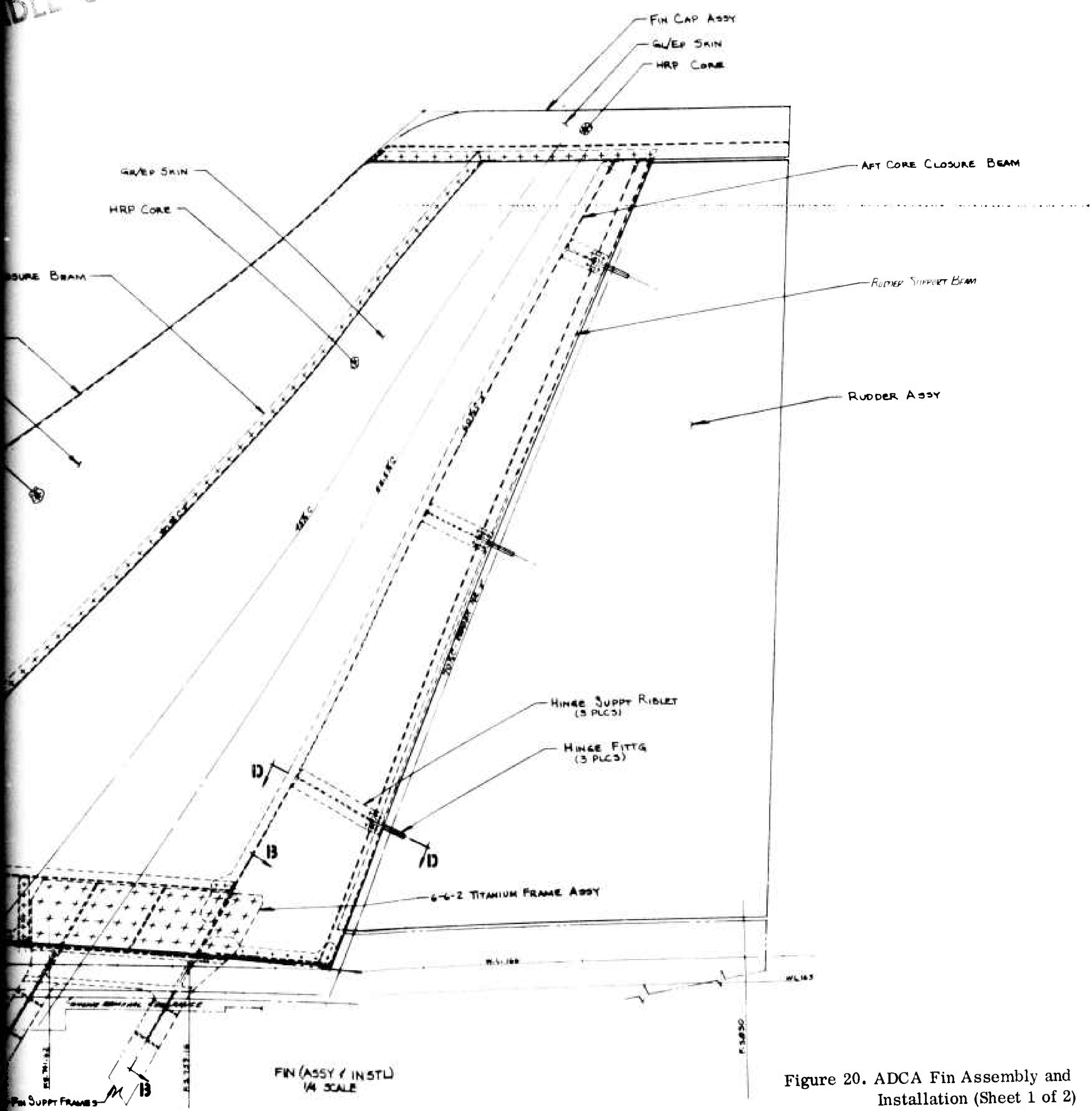
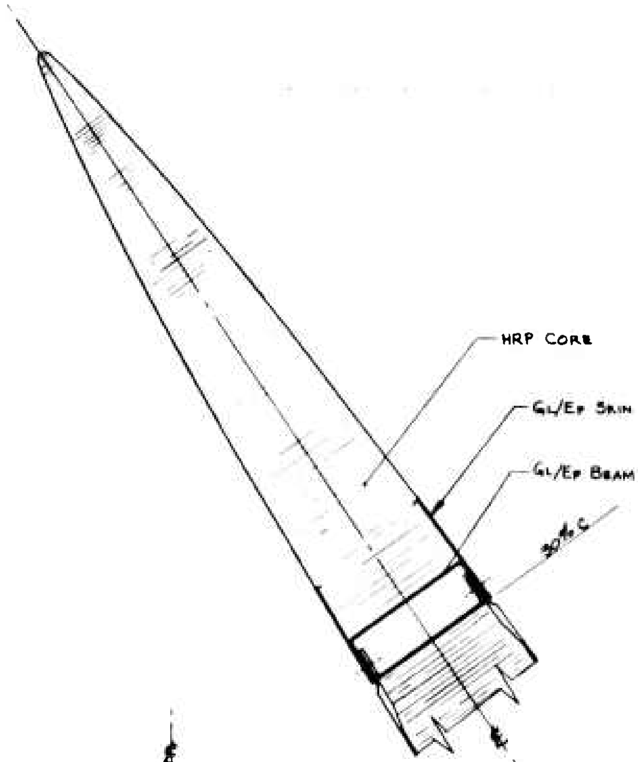
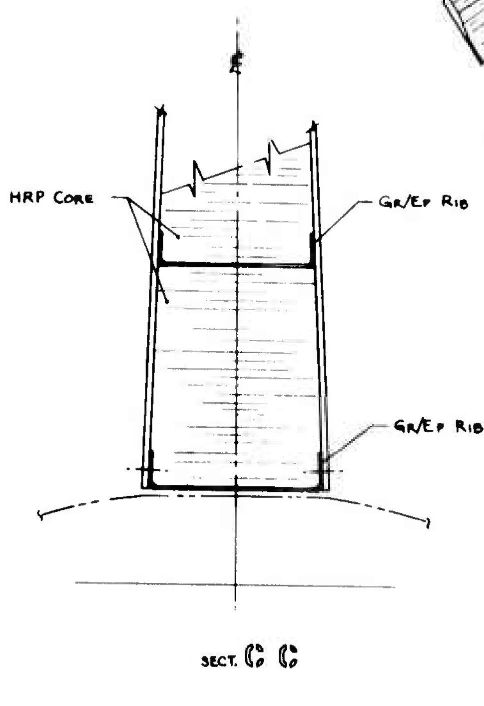


Figure 20. ADCA Fin Assembly and Installation (Sheet 1 of 2)



SECT. A-A
HALF SCALE



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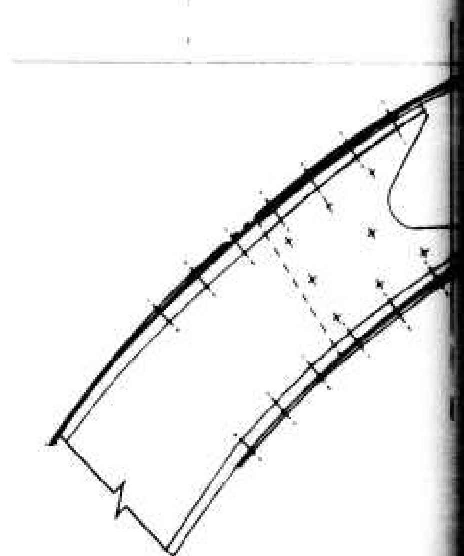
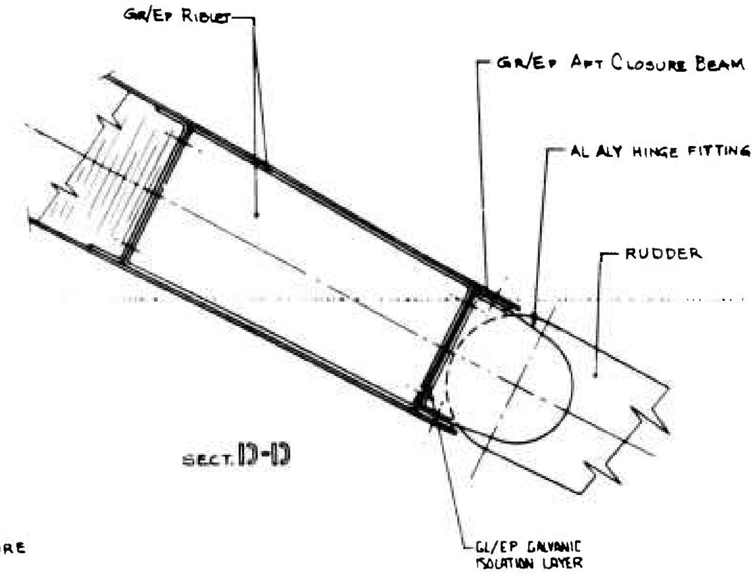




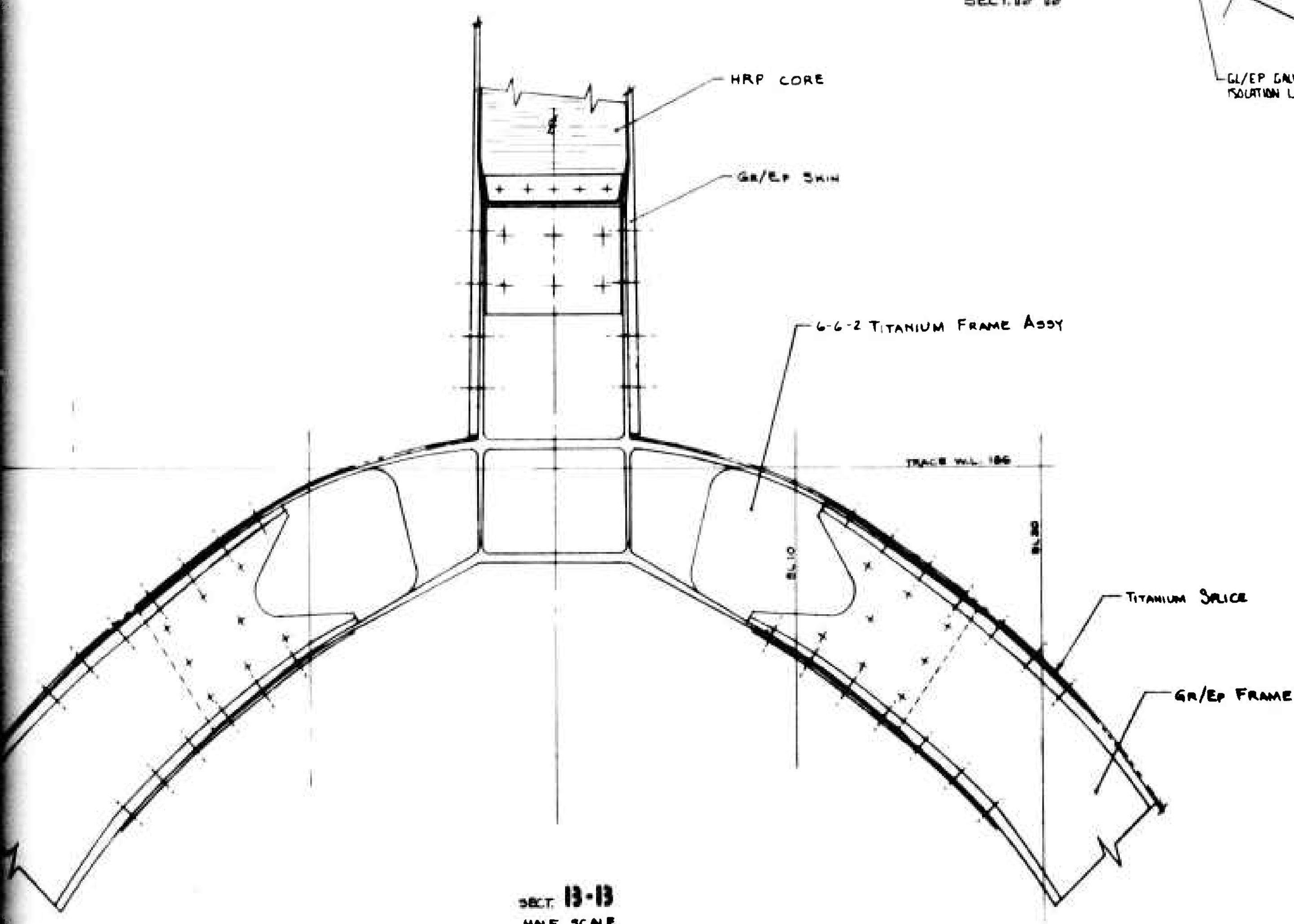


Figure 20. ADCA Fin Assembly and Installation (Sheet 2 of 2)

The assembly is mounted on chordline in a holding fixture and the core machined using a N/C tape derived from the skin mold. After verifying the bond line using the adhesive isolation technique, the skins are bonded to the substructure. The bonding fixture has a diaphragm face to provide uniform bonding pressure despite skin thickness variations.

5.2.2 Vertical Fin Final Assembly

The full-bonded torque box is mounted in an assembly fixture and the cover-to-hinge support riblet joints are made using blind fasteners. Then the leading edge and tip cap are attached, and the joints sealed. The rudder is prefitted to the vertical fin assembly, the hinge attachments are then made, and the two components are disassembled and sent to the final assembly operation.

5.3 CANARD FABRICATION

The canard is a sandwich structure with Gr/Ep ribs closed out with a Gr/Ep leading edge and Gl/Ep tip caps (Figure 21). A welded steel box fitting provides for attachment to the fuselage. Integrally molded stepped titanium splice plates in the skins are used to provide the load path into the structure. The critical upper skin and substructure parts are precured. The lower skin is cocured as the assembly is bonded.

5.3.1 Fabrication of the Canard Torque Box

The upper cover skin mold is a core faying design and provides for coordination of the chem milled titanium splice plate. A peel ply is laid up on the mold followed by half the skin plies and the layup is debulked. Concurrently, the splice plate is pretreated and the adhesive laid up where required. The splice plate is located on the tool and the fiberglass transition plies as laid up. The remaining skin plies are laid up by the laminating center and debulked. The skin assembly is then flame-sprayed and the bleeder system installed. After autoclave cure and oven post-cure, the skin is inspected.

The Gr/Ep ribs are prepared as detail parts on male tools. The Gl/Ep trailing edge is press-molded in mated dies using a tapered layup. The core details are cut to net plan-form dimension, but oversize in thickness, and attached to a tooling plate with polyethylene glycol. The core surface next to the precured skin is then machined to final dimension.

The bonding fixture has a diaphragm face and side rails to locate the ribs and closure members. A jigged shaft is provided to locate the shaft fitting. The detail parts are loaded in the fixture and prefitted. The core details are then attached to the precured skin with doubled-back masking tape and machined to contour. The detail parts are removed from the fixture and prepared for bonding. After the adhesive is laid-up on the parts, the cured skin and substructure is reinstalled in the bonding fixture. The Gl/Ep overwrap at the trailing edge is laid-up on the precured skin and subsequently folded over the lower skin layup. The substructure is covered with film adhesive and a layer of precured fiberglass is laid-up over the core. A second layer of adhesive is laid-up over the fiberglass. Next, the lower skin and splice plate is laid-up and flame-sprayed in the same sequence used for the upper skin. The assembly is bagged and autoclave-cured under reduced pressure to cocure the lower skin and form all the bonds. The bonded assembly is postcured at 350° F, and the net trim edges sanded to remove resin flash. After inspection, the bonded assembly is ready for final assembly.

As an alternate process, secondary bonding of precured skins will be considered since two skin molds will be available. This manufacturing concept trades off manufacturing cost against weight reduction and increased inspectability. If the canard were fabricated by secondary bonding, the processing would be similar to that of the vertical fin, Subsection 5.2.

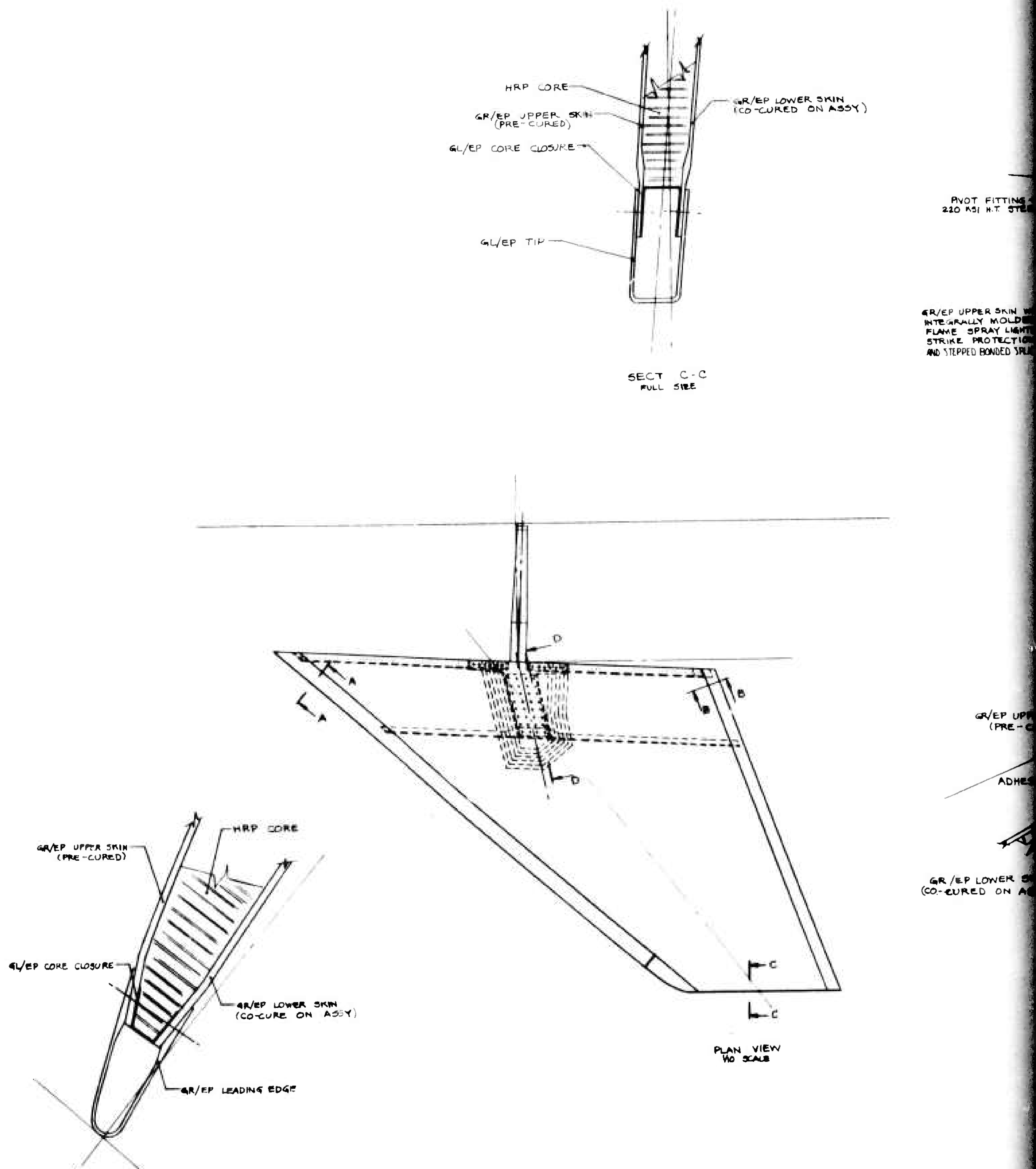
5.3.2 Canard Final Assembly

The sandwich panel is mounted on the chord plane in an assembly fixture and the splice plate-to-pivot fitting fastener holes are drilled/countersunk using a Quackenbush drill with a cobalt cutter. Next, titanium Huckbolt fasteners are installed with wet sealant. The leading edge and tip are then prefitted to the assembly, drilled, and attached with Jo-bolts. All bolt heads, attachment areas, and exposed edges are sealed with polysulfide sealant. After inspection, the canard is ready for final assembly.

5.4 FUSELAGE FABRICATION

The fuselage, Figure 22, is of semi-monocoque construction comprising 17 major frames and bulkheads, a structural outer shell and a sandwich inlet duct. The outer shell contains several integrally molded longeron sections and two upper longeron detail parts, and serves as a pressure vessel in the areas of the cockpit and mid-module integral fuel tank. The last mid-module bay also contains a bladder fuel tank.

The cockpit floor and tank deck are structural and tie the bulkheads and outer shell together. The longerons are spliced as needed by composite plates. Titanium fittings, attached to the bulkheads, are used to install the wing assembly, vertical fin, canards, and landing gears.


HRP CORE
GR/EP UPPER SKIN
(PRE-CURED)
GL/EP CORE CLOSURE
GL/EP TIP
GR/EP LOWER SKIN
(CO-CURED ON ASSY)
SECT C-C
FULL SIZE
PIVOT FITTING
220 KSI H.T. STE
GR/EP UPPER SKIN W
INTEGRALLY MOLDE
FLAME SPRAY LIGHT
STRIKE PROTECTIO
AND STEPPED BONDED SPL
GR/EP UP
(PRE-C
ADHES
GR /EP LOWER S
(CO-CURED ON A
PLAN VIEW
1/10 SCALE
GR/EP UPPER SKIN
(PRE-CURED)
HRP CORE
GL/EP CORE CLOSURE
GR/EP LOWER SKIN
(CO-CURE ON ASSY)
GR/EP LEADING EDGE
SECT A-A
FULL SIZE

Figur

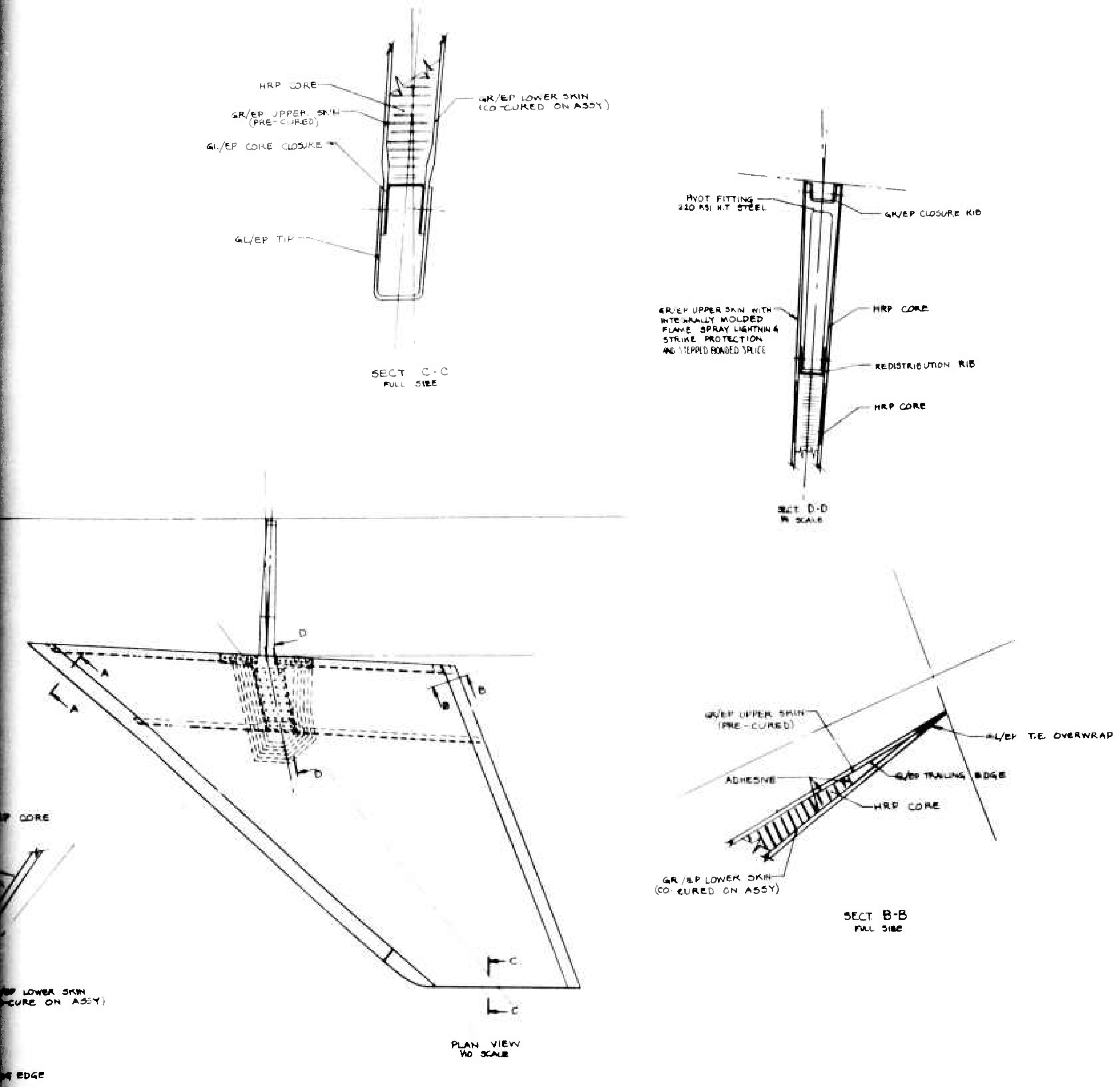
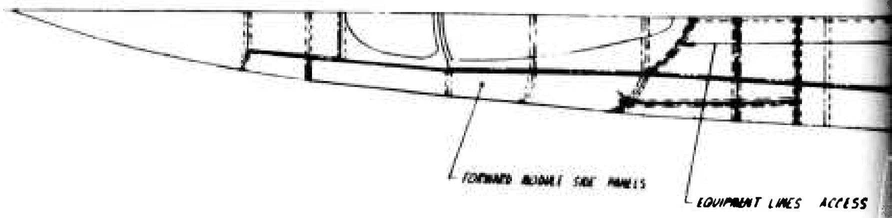
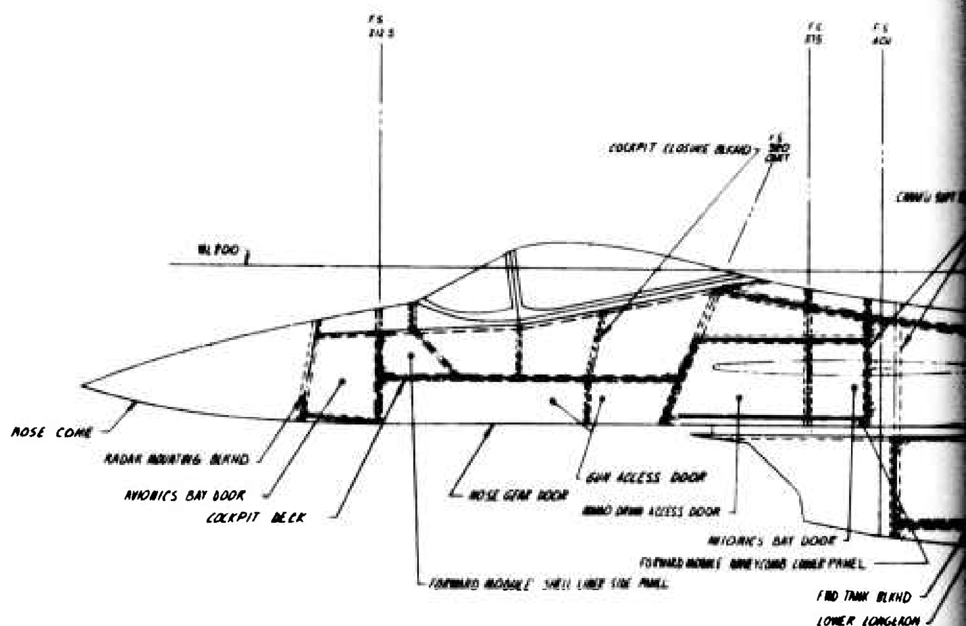


Figure 21. ADCA Canard Structural Concepts


FORWARD MODULE SIDE PANELS
EQUIPMENT LINES ACCESS


COCKPIT CLOSURE BLKHD
WL 200
NOSE CONE
RADAR MOUNTING BLKHD
AVIONICS BAY DOOR
COCKPIT DECK
NOSE GEAR DOOR
GUN ACCESS DOOR
AVIONICS BAY DOOR
FORWARD MODULE SHELL LINER SIDE PANEL
FWD TANK BLKHD
LOWER LONGERON

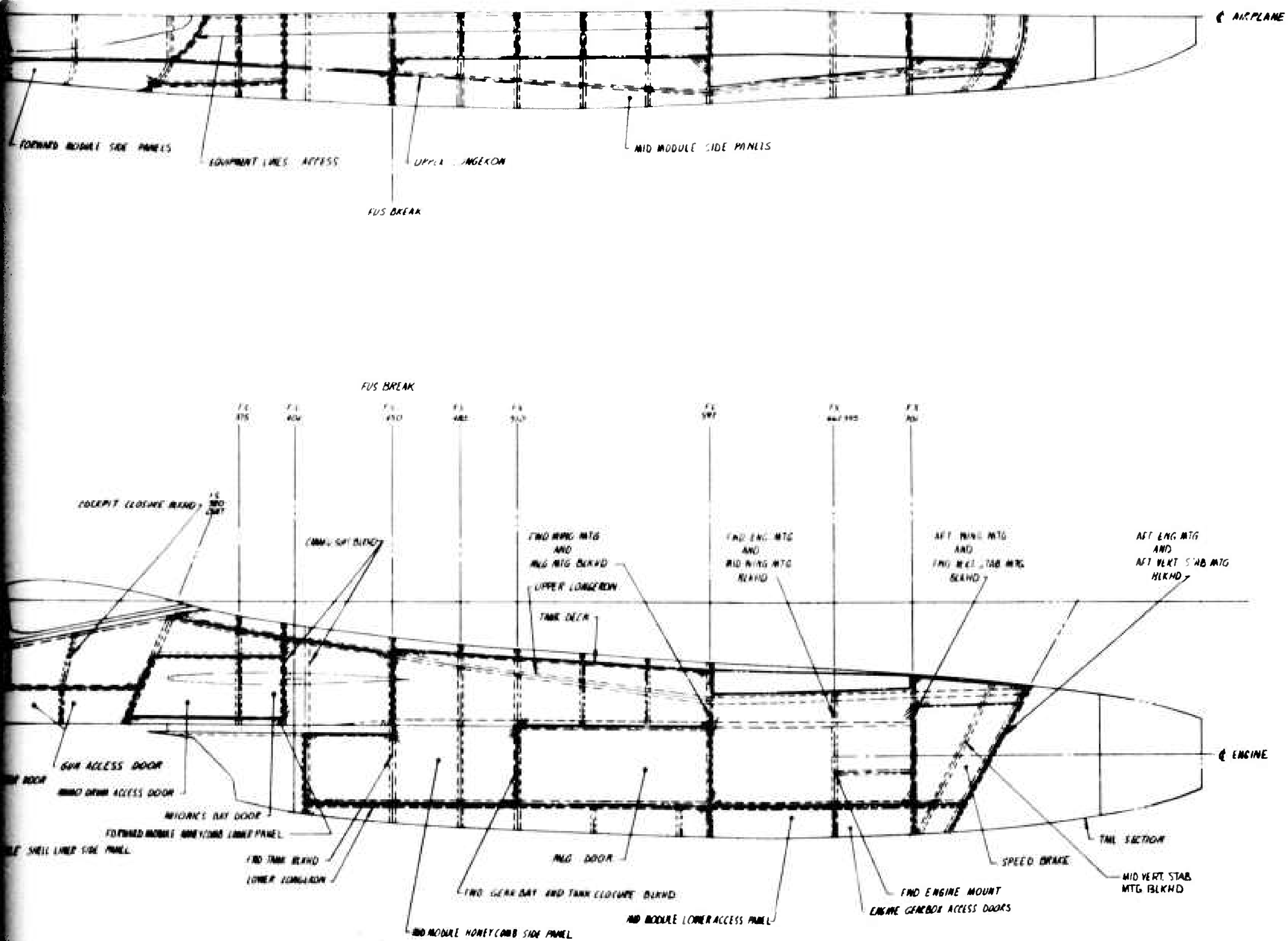


Figure 22. ADCA Fuselage Structural Concept (Sheet 1 of 4)

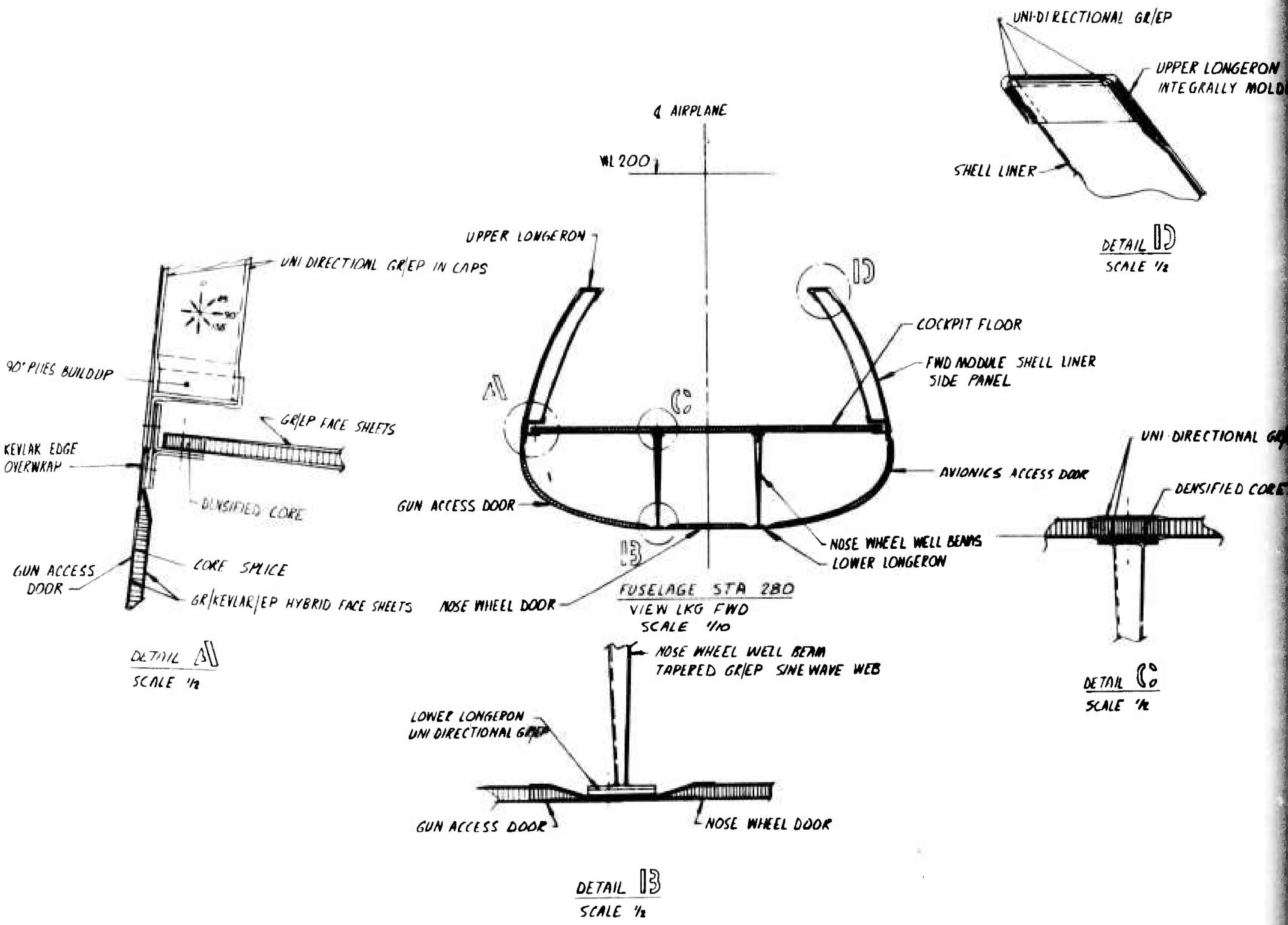


Figure 22. ADCA Fuselage Structural Conc
(Sheet 2 of 4)

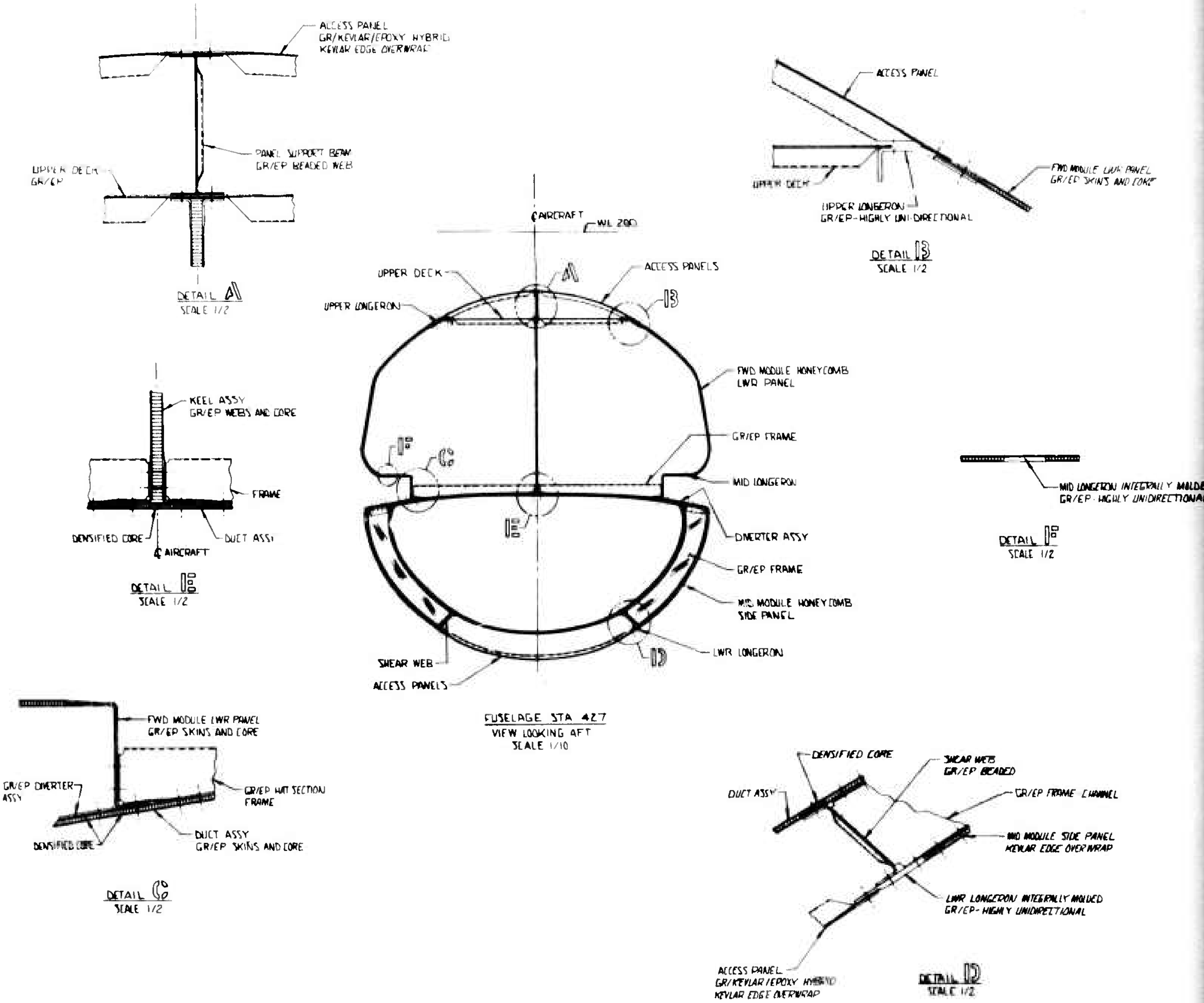


Figure 22. ADCA Fuselage Structural Concep
(Sheet 3 of 4)

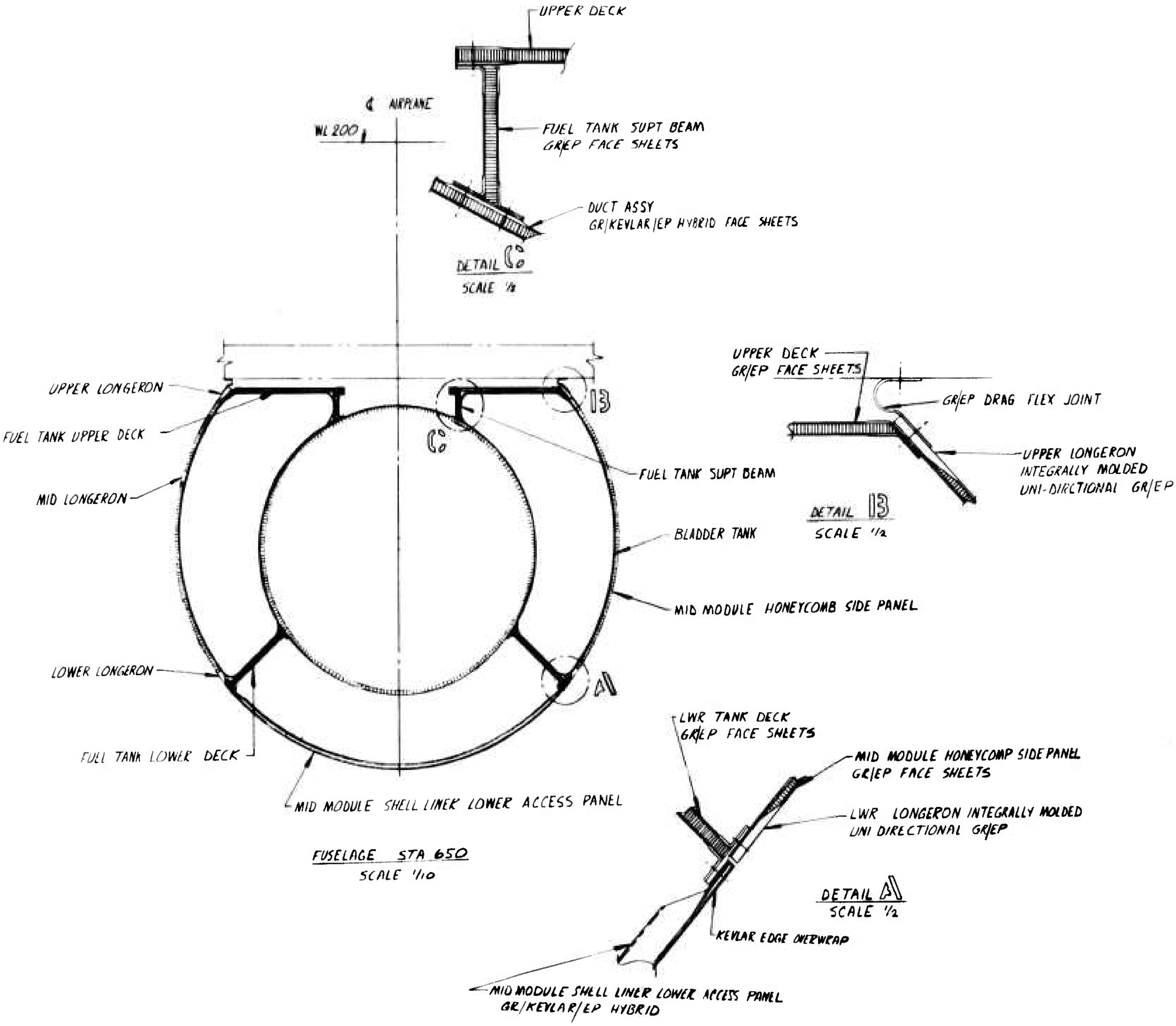


Figure 22. ADCA Fuselage Structural Concept
(Sheet 4 of 4)

The fuselage consists of four major components: the forward module, the mid module, the aft module (tail cone) and the two-piece duct assembly. The major fuselage splice occurs at the station 450 bulkhead which closes out the mid module. This arrangement permits sealing and checkout of the internal fuel tank prior to joining the forward module.

The primary structural composite material is graphite/epoxy with Kevlar/epoxy used in access doors to reduce the possibility of handling damage. Integrally molded foil inserts are used in areas where doors are frequently opened. Kevlar/epoxy is also hybridized with Gr/Ep in the ducts.

5.4.1 Forward and Mid Module Detail Part Fabrication

The various types of detail parts in these modules can be classified into the 10 basic types discussed below. Wherever possible, they are produced by net trim cocure molding processes. The individual plies are laid up and trimmed by the laminating center or blanked from preplied sheet prepregs as applicable. The laminating center, web corrugating machine, or other mechanical devices are used to stack and form the plies on the tool. Hand layup and stacking are used only when automation is not feasible.

5.4.1.1 Sandwich Bulkhead

The sandwich bulkheads, typified by the station 663 bulkhead, Figure 25, are attached to the duct by composite angles, and are flanged to accept the skin panels. These parts will be cocured using rigid, split steel female tools to control the outer and inner cap contours and prevent core crushing. The web surfaces will be controlled by two contoured, rubber faced metal plugs which apply pressure. The plugs are coordinated by pins located in cutout areas of the bulkhead. Thin aluminum caul plates are used over the core in the webs to prevent dimpling. During skin curing, the core assembly is spliced, closed-out with low-density potting compound, and environmentally sealed with two plies of Gl/Ep at the inner diameter. Precured Gl/Ep details are used to close-out the core where attachment fittings are inserted between the skins. During cure, this cavity is filled with a machined steel plate. The layup sequence is:

- Stack and form channel sections on plugs with caul plates and bleeders in place
- Layup core assembly on lower plug/layup assembly
- Install cavity forming plate
- Install upper plug/layup assembly

- Layup radius adhesive
- Layup cap plies, trim layup, and install cap bleeder
- Install split female cap tools and join segments
- Apply vacuum bag

5.4.1.2 All-Composite Bulkhead or Frames

The all composite bulkheads are typically of channel or I cross-section, depending on the load intensity. The bulkhead at FS703, Figure 25, has two cavities to accept the aftwing attachment fittings. The web of one channel is extended as a flat plate between the two fittings to provide for secondary attachment to the wing. The parts are cured on matching rubber faced metal plugs coordinated by pins located in cutout areas of the bulkhead, Figure 23.

A thin caul plate and local silicone rubber pressure intensifiers, as needed, are used to form the cap members. A Teflon-coated steel plate is inserted in the layup to produce the cavity required for the fitting. The layup follows the sequence used for the sandwich bulkheads, except that no core is used and then the cap caul plates replace the split steel molds.

5.4.1.3 Stiffened Skin Bulkhead

The stiffened skin bulkheads, such as the radar mounting bulkhead, are channel cros-section with integrally molded beads or hat sections. The beaded web bulkheads will be molded on contoured male molds with a preformed silicone rubber pressure pad to ensure uniform application of curing pressure. The frame with hat section stiffiners will be processed as described in Subsection 4.3.1 using the tooling concept shown in Figure 12. Both types of parts will be net molded to eliminate trimming.

5.4.1.4 Sandwich Skin Panel

Sandwich skin panels, such as the mid-module side panel, (Figure 22, sheet 3) also contain longerons. In order to guarantee longeron stiffness, the air passage skin will be precured and fully inspected. The inner skin and core closeouts will then be cocured. The manufacturing risk is too great for one-shot cocuring. The molds are of air passage contour eggcrate design fitted with pressure bars to prevent crushing the prestabilized ramps in the core. Contoured silicone rubber pressure pads or vacuum bags are used

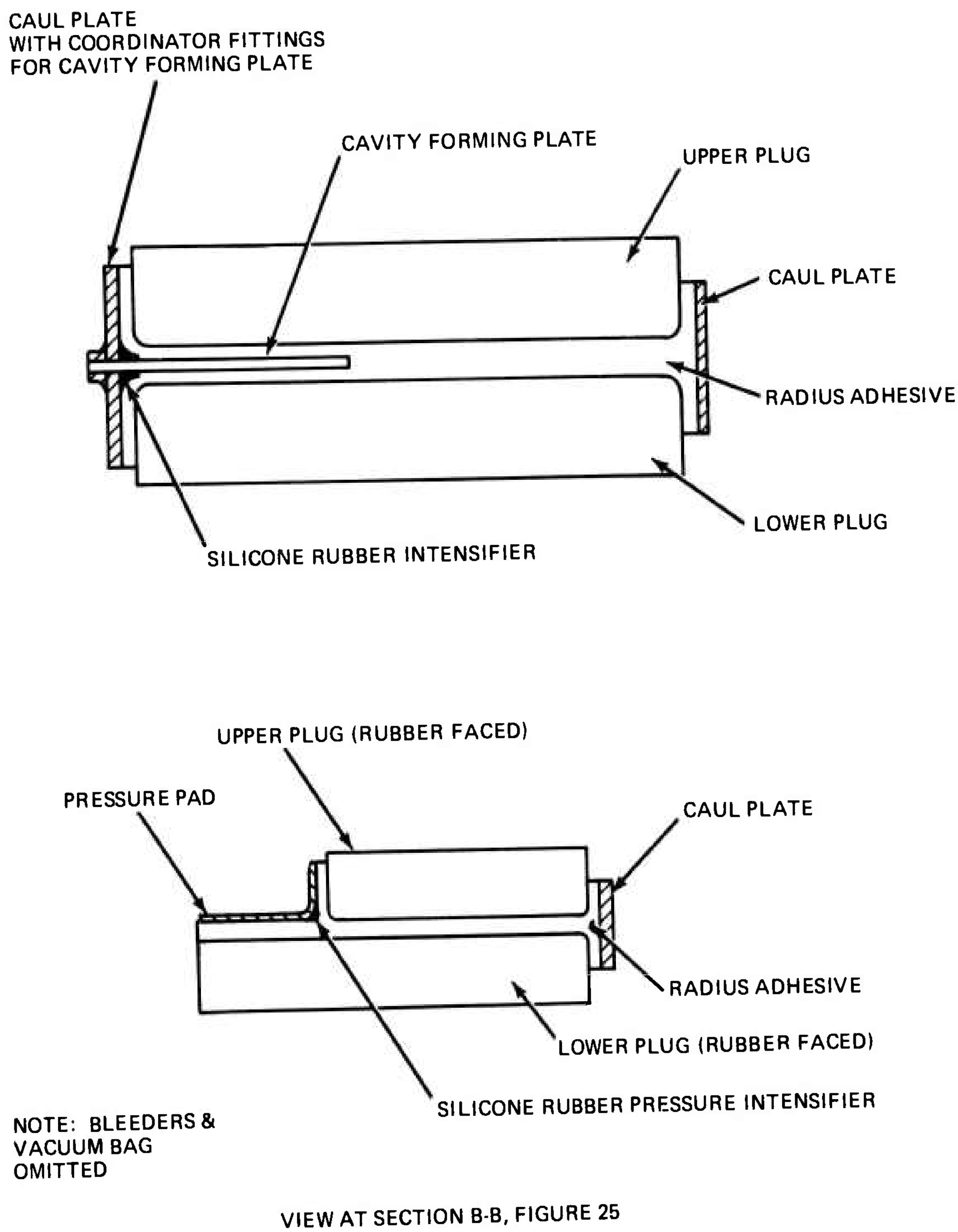


Figure 23. All-Composite Bulkhead Tooling Concept

for uniform pressure distribution. The Kevlar/epoxy edge wraps are made as part of the cocuring process for the inner skin.

5.4.1.5 Shell Liner Skin Panels

These panels, such as the forward module side panel (Figure 22, sheet 2), are processed in the same manner as the hat section stiffened bulkheads, Subsection 5.4.1.3. The layup sequence is:

- Layup of air passage skin with upper longeron plies protruding beyond the skin trim.
- Layup of stiffener/mandrel assemblies
- Folding of upper longeron plies over edge of skin and top of stiffener/mandrel assemblies

A preformed pressure pad, in conjunction with local pressure intensifiers, is used to prevent molding defects. The mandrels are removed through openings at the lower end of the hat sections.

5.4.1.6 Sandwich Deck Panel

The cockpit floor panel (Figure 22, sheet 2) and fuel tank deck panel contain a locally reinforced, constant thickness honeycomb blanket sealed with two plies of Glass/Epoxy material. The skin thickness variations are reflected away from the core. The core blanket is prefabricated using a steel frame with face plates during vacuum bag cure of the splice adhesives and potting compound. The same frame is used to cocure the skins.

5.4.1.7 Sandwich Support Structure

The internal sandwich support structure is cocured using the process and tooling plan established for the sandwich bulkheads, Subsection 5.4.1.1. The only modification is that a base plate is used to tie the tooling together at the edges of the part.

5.4.1.8 Tapered Sinewave Spar

The nosewell wheel beams (Figure 22, sheet 2) are a modification of the B-1 Composite Horizontal Stabilizer sinewave spar design. They are produced by:

- Forming two channel sections on padded matching metal tools
- Joining the two layup/tool assemblies
- Installing radius adhesive

- Laying-up cap laminate assemblies and net-trimming the layup
- Installing cap bleeder system and caul plates
- Curing by the envelope bag process

The channel sections will be formed by corrugating flat layups of prepreg and bleeders onto the web areas of the tools, followed by vacuum-forming the flanges with a precontoured rubber bag. As an alternate, the flange next to the flat section of the web can be formed over a block, and the layup/bleeder assembly placed on the tool. Then the corrugated web regions and associated flange are formed by hand before the layup is vacuum compacted.

5.4.1.9 Beaded Gr/Ep Shear Webs

The beaded Gr/Ep shear webs (Figure 22, sheet 3) are molded on male steel tools using a precured silicone rubber pressure pad.

5.4.1.10 Upper Longeron

The upper longerons (Figure 22, sheet 3) are laid up from preplied sheet and molded on steel male tools. A preformed pressure pad is used to prevent radius wrinkles. The parts are net molded slightly oversized, and the edges are trimmed by routing using a template.

5.4.2 Aft Module (Tail Cone) Fabrication

The tail cone assembly is a honeycomb sandwich structure closed out with circular channel sections to provide for attachment to the conical aft engine bulkhead and metallic exhaust cone. The small part diameter-to-length ratio makes layup in female air passage tooling very difficult and costly. The contoured closeout of FS650 and conical shape of the structure are not adaptable to one-shot cocuring. Therefore, the closeout members and inner skin will be precured, and the outer skin cocured. The major tool is a male plug on which the assembly is built up. The plug will be fitted with rails to position the channels and prevent core crushing. The plug also serves as a mandrel to rotate the assembly during core machining, if required. The conical skins are laid-up on full ply mylars by the laminating center and hand-stacked on the tool. The individual plies are overlapped at the joint, and the seams are staggered. The core details are preformed to match the inner skin, preferably at net thickness if at all possible.

The fabrication flow is as follows:

- Layup, cure, and inspect inner skin
- Layup, cure, and inspect channel members
- Prepare detail parts for bonding
- Replace inner skin on plug and lay-up adhesive
- Install channel sections/tooling assemblies at proper locations
- Layup core details, core splice adhesive, and potting compound as required
- Install segmented, dummy outer skin
- Cure at moderate autoclave pressures
- Machine core to final dimensions (if required)
- Inspect bonds and repair core/skin bonds
- Layup one-ply film adhesive over core
- Layup segmented, contoured, single ply Gl/Ep sheet
- Layup second ply film adhesive and composite skin plies
- Install bleeder system and nylon film vacuum bag
- Autoclave cure at reduced pressure
- Inspect and final trim at ends

This fabrication method has been verified by the successful test of a C-4 missile equipment bay adaptor approximately 72 in. in diameter and 36 in. high.

5.4.3 Inlet Duct Fabrication

The sandwich inlet duct and associated fairing extend from FS 350 to approximately FS 740. The duct geometry changes from elliptical to a circular cross-section. The duct is split at FS 450 so that the two sections can be inserted in cutouts in the fuselage bulkheads. The splice is made using an external band attached by bonding and fastening. The skins are locally padded in high load attachment areas, and the core is densified with low-density potting compound as required.

The critical surface for preventing turbulance in the duct is the inner mold line. Slight tolerance accumulations in the external surface are accommodated by adjusting the clips which attach the duct to the bulkheads. The duct geometry precludes the use of female tools or secondary bonding processes. The duct assemblies will therefore be fabricated using the sequences described for the tail cone, subsection 5.4.2.

The exact duct contour has not been fully defined under this study program, but the changes in cross-section are probably large enough to require modification of the layup process. These modifications might include:

- Use of 3D layup templates with tapered Gr/Ep tape strips
- Laying tape ply on ply on the tool with special tooling to correct the orientation periodically
- Cut and tuck layup of woven Gr/Ep prepregs

The woven Kevlar/epoxy plies are laid up using the cut and tuck process and internal ply splices as directed by the engineering drawing.

5.4.4 Forward Module Structural Assembly

The forward module assembly plan reflects the composite design philosophy of significantly reducing the part and fastener count from those found in a metal airplane. As a result, the module is assembled in a single fixture rather than joining several subassemblies. Higher rates are accommodated by duplicating this fixture and mounting the partially assembled modules in dollies for system stuffing and prefitting the doors.

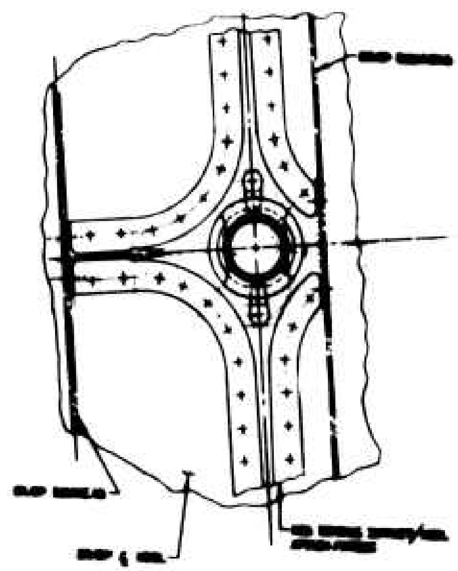
The structure is assembled in three stages. First, the cockpit deck, upper deck, keel assembly, forward duct section, and all bulkheads forward of FS 410 are mounted in the assembly fixture. Part location is controlled by tooling holes drilled into the various parts prior to removal from the bonding fixture. The joints between the various bulkheads and the decks or duct, are made using either clips or integrally molded flanges. The various parts are prefitted and liquid-shimmed as required. Installation fixtures, coordinated to the assembly fixture, are used to control the location of fastener holes. Then, the wheel well beams are installed. The canard inboard bearing support/keel attachment fitting, Figure 24, is attached to the keel, and the forward landing gear attachment fitting is mated to the nose gear mounting bulkhead. A hoist fitting is attached to the FS 407 bulkhead to complete assembly of the internal structure.

If desired, the complete accessibility of the module can be used to install cables, hydraulic lines, and associated brackets before the fixed skin panels. The right-hand and left-hand lower side panels, and Gr/Ep fuselage frame (Figure 22, sheet 3), are drilled and installed with temporary hardware. They are subsequently removed to provide access during mating of the forward and mid modules during final assembly. Then, the shell liner side panels are permanently attached, and the upper access panel installed. The canopy attachment fitting is fastened to the module, and the canopy attached. The cockpit area is sealed and pressure-tested. Then the access doors are prefitted and drilled to complete the forward module assembly.

5.4.5 Mid Module Final Assembly

The mid module assembly plan is similar to that for the forward module since the fuel tank deck and duct are used to tie together the bulkheads before the various internal beams are attached. Numerous metal fittings are fastened to the mid module bulkheads for attaching other structure and the engine. Six titanium fittings provide for joining the fuselage to the wing (Figure 25). Steel landing gear support and drag fittings are fastened to the bulkhead at FS-600 (Figure 26). Attachment of the vertical fin is also provided by fittings joined to the mid module (Figure 20). During the assembly sequence, all joints are sealed before they are closed out since the forward bays of the mid module are an integral fuel tank. The mid module is skinned by three major skins. The right-hand and left-hand side panels are 25 ft. long honeycomb panels containing integrally molded upper, mid, and lower longerons. These panels tie into the last bulkhead of the forward module in the main fuselage splice. The aft upper panel also is a sandwich construction. These three skins are prefitted to the substructure and drilled. Bladder fuel tanks in the engine area and associated systems, as well as electrical cables and hydraulic lines, are installed prior attachment of the skins.

The various access doors and speed brakes are installed and the integral fuel tank bays are pressure-checked to complete the assembly process.



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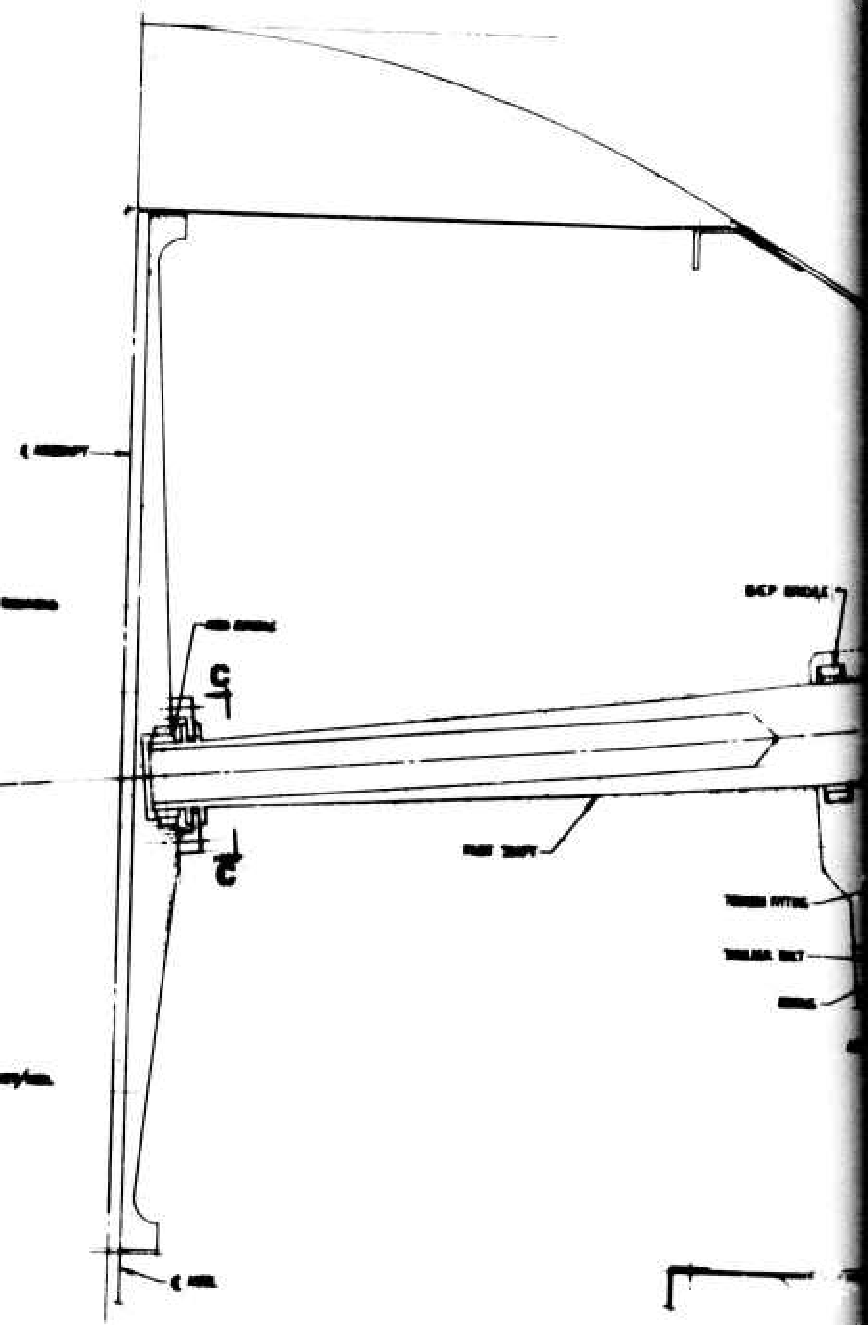

SECTION THRU ℄ OF PIVOT SHAFT

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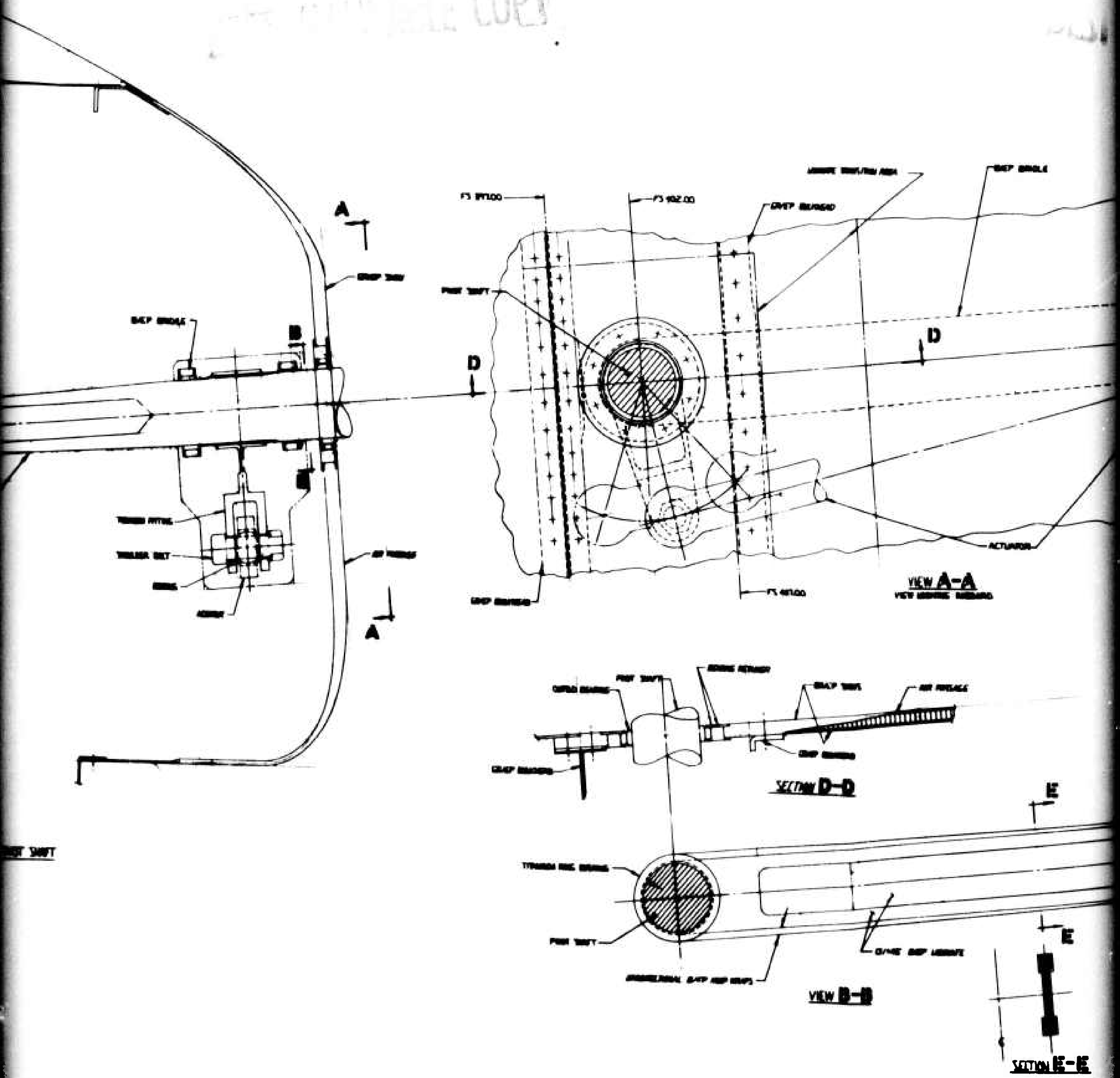
A
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ACTUATOR
VIEW A-A
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SECTION E-E

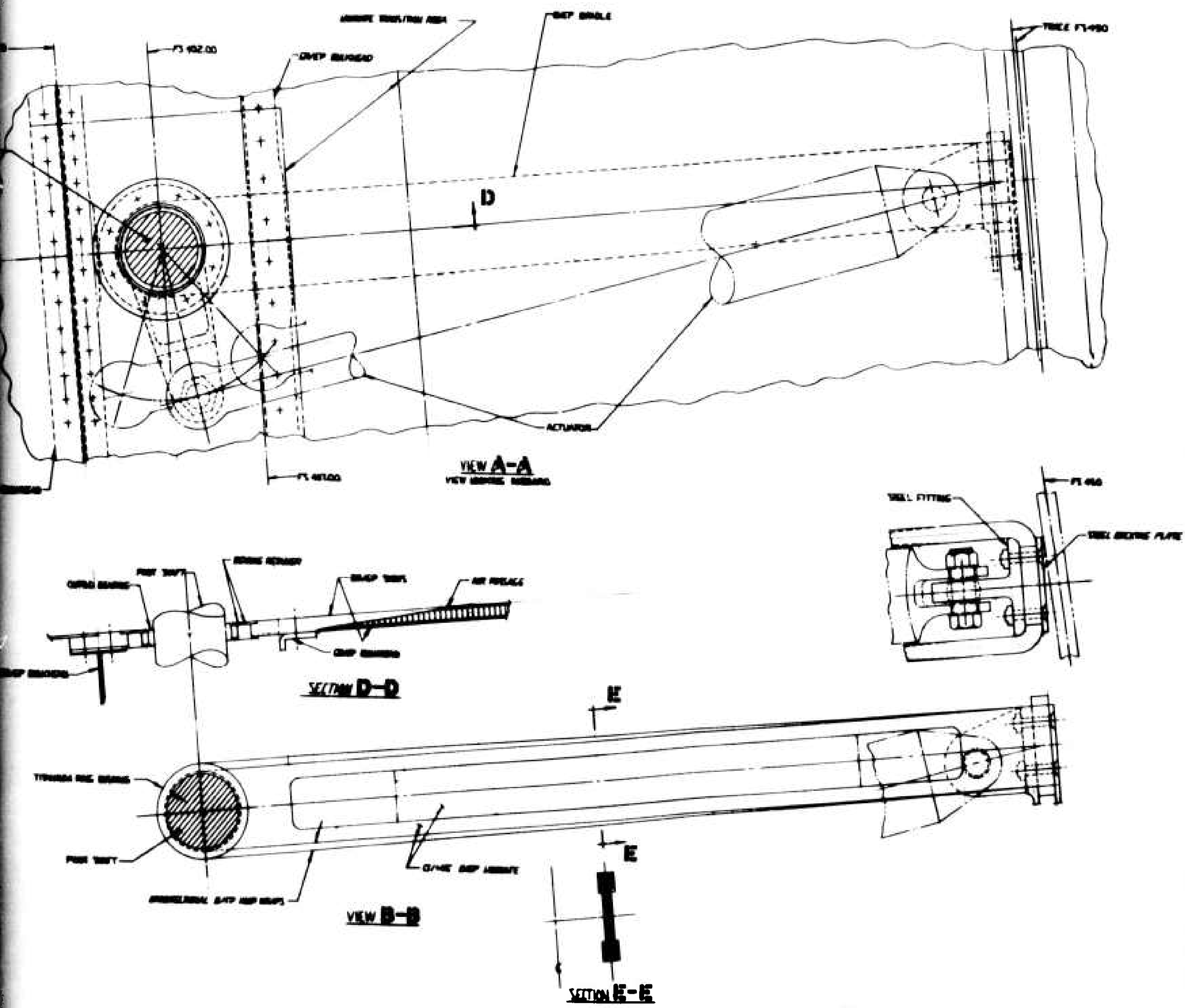
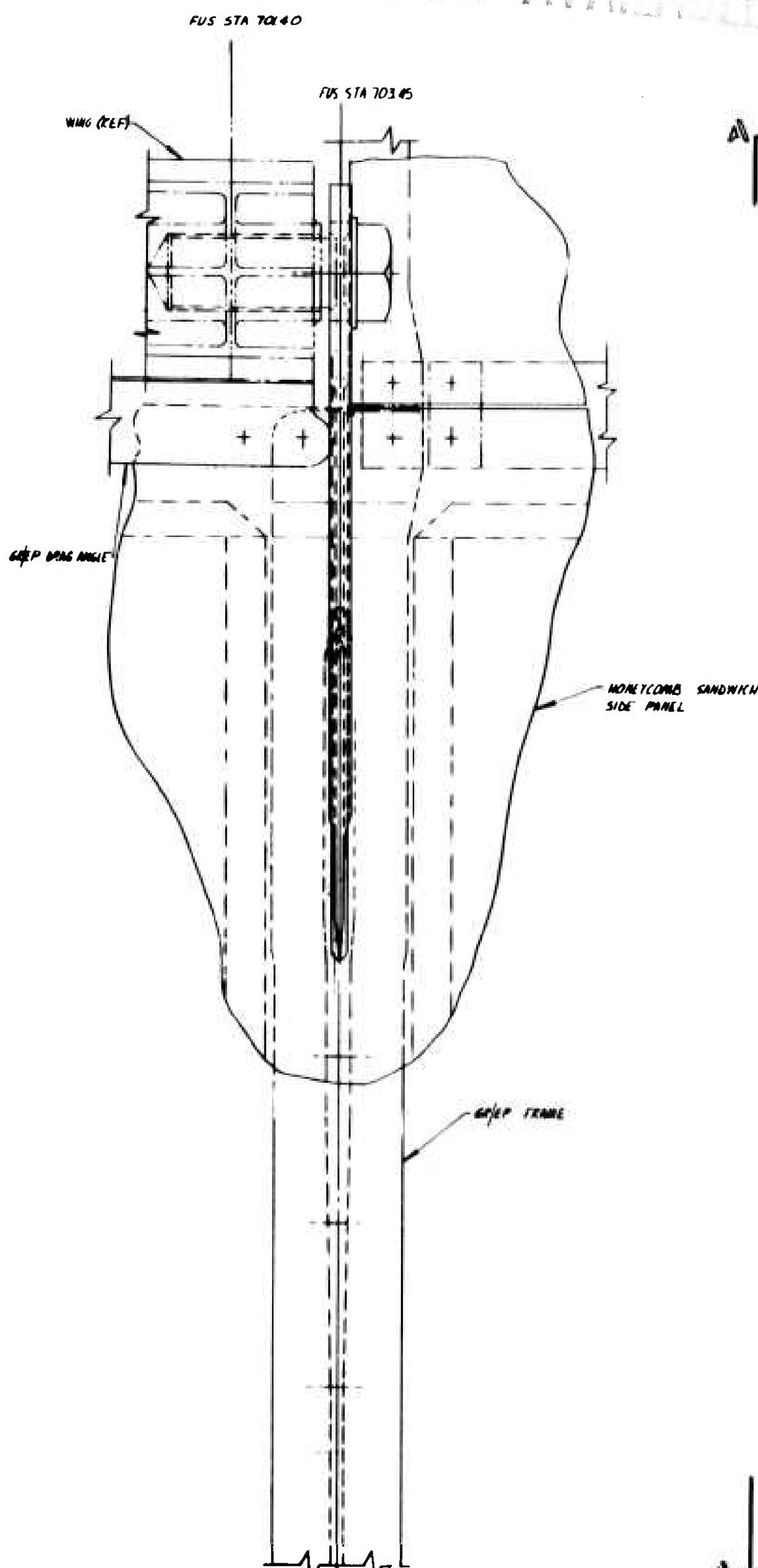


Figure 24. ADCA Canard/Fuselage Attachment



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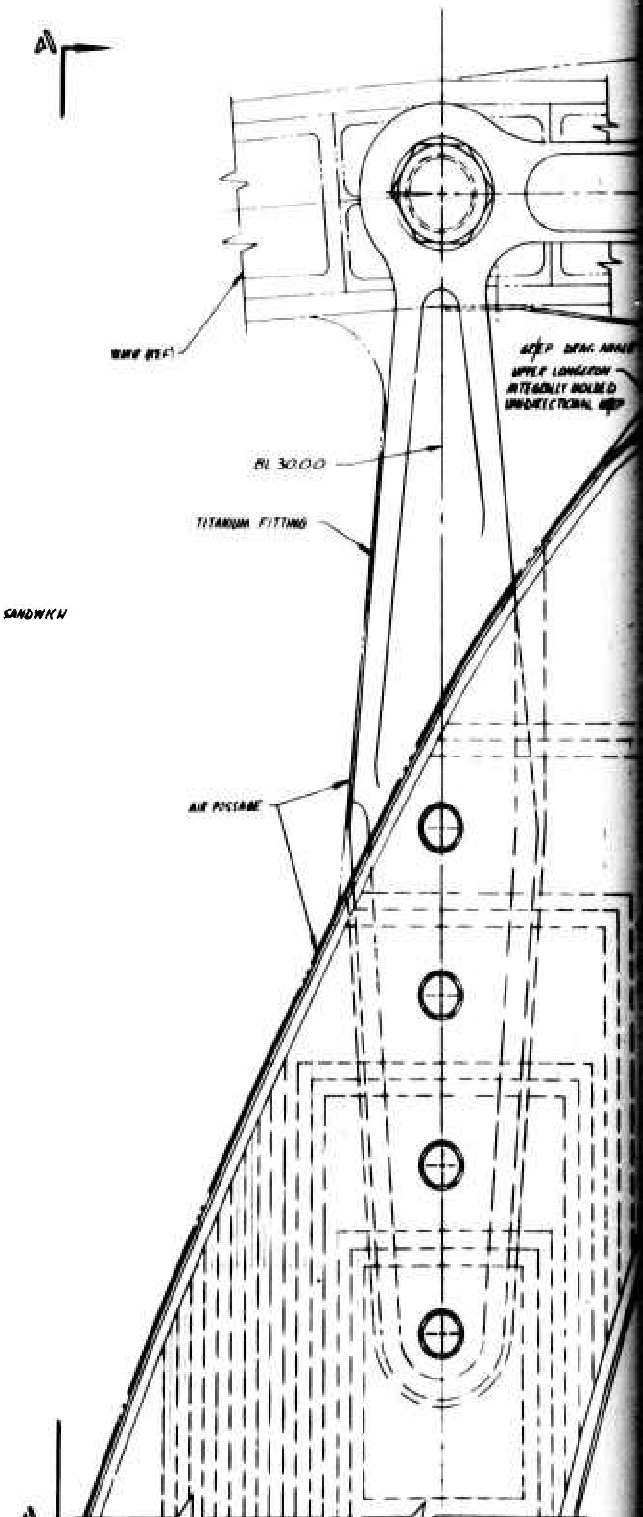

VIEW LKG FWD
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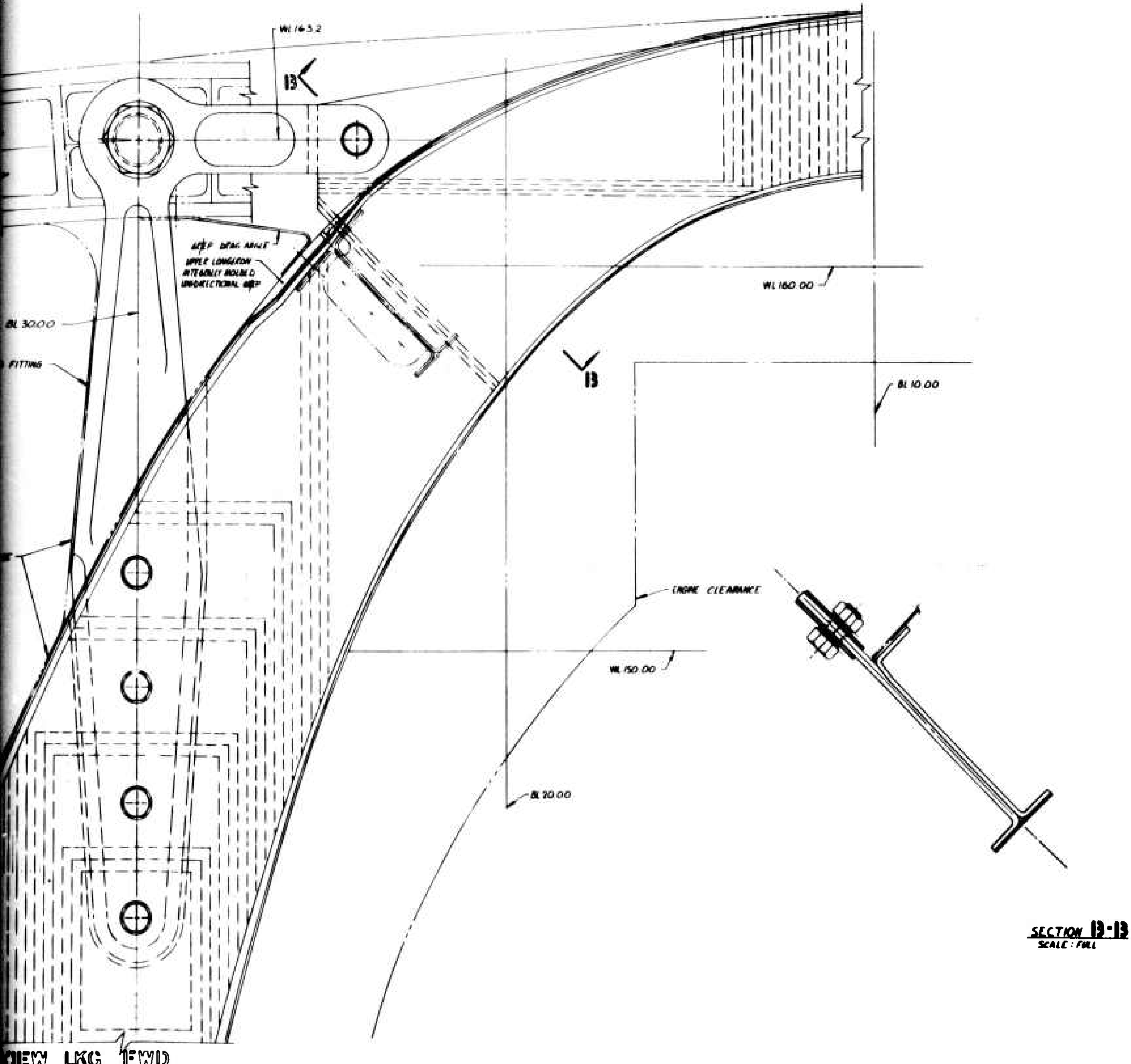
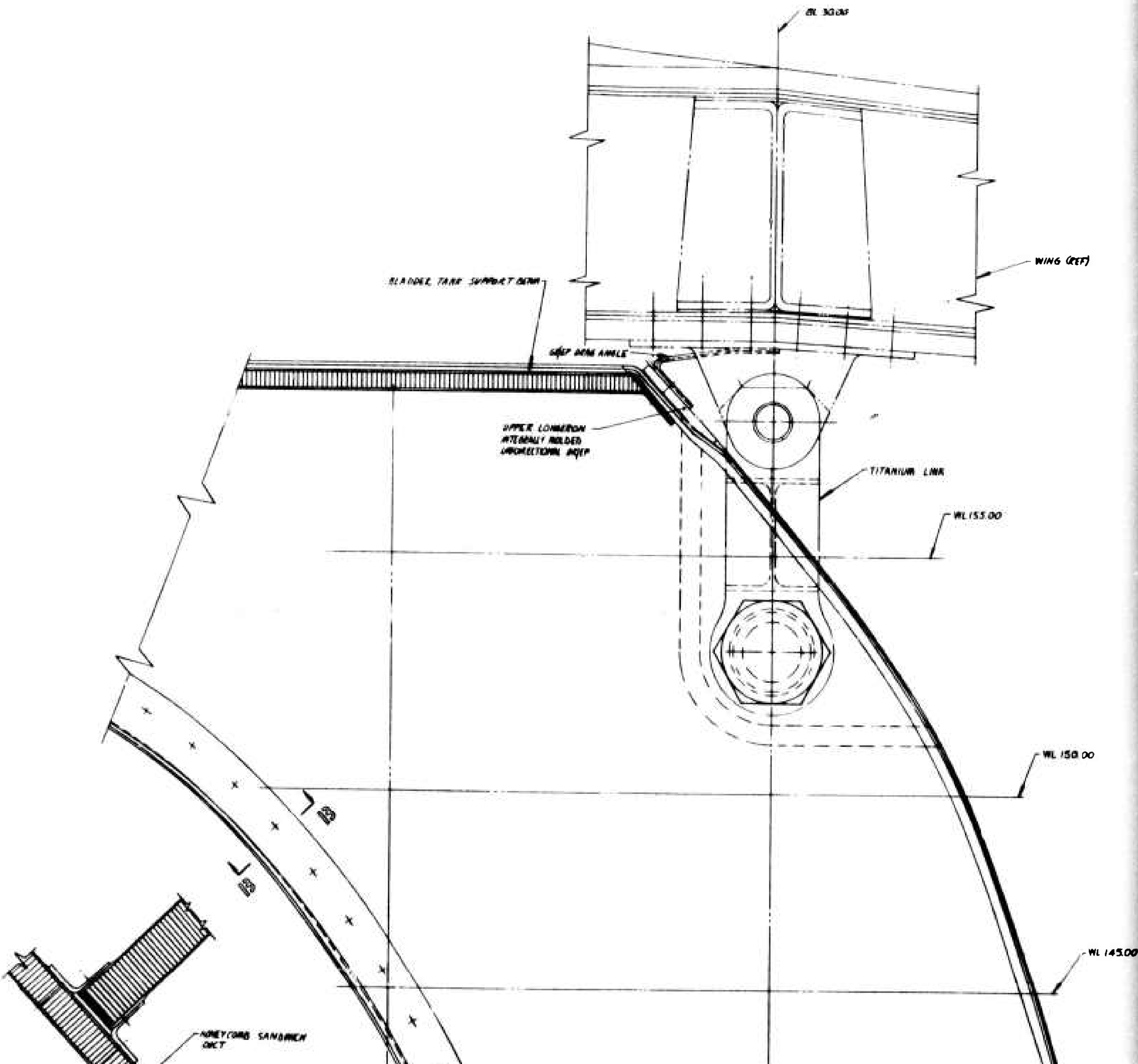


Figure 25. ADCA Wing/Fuselage Attachment (Sheet 1 of 2)



SECT 13-13
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VIEW LKG AFT
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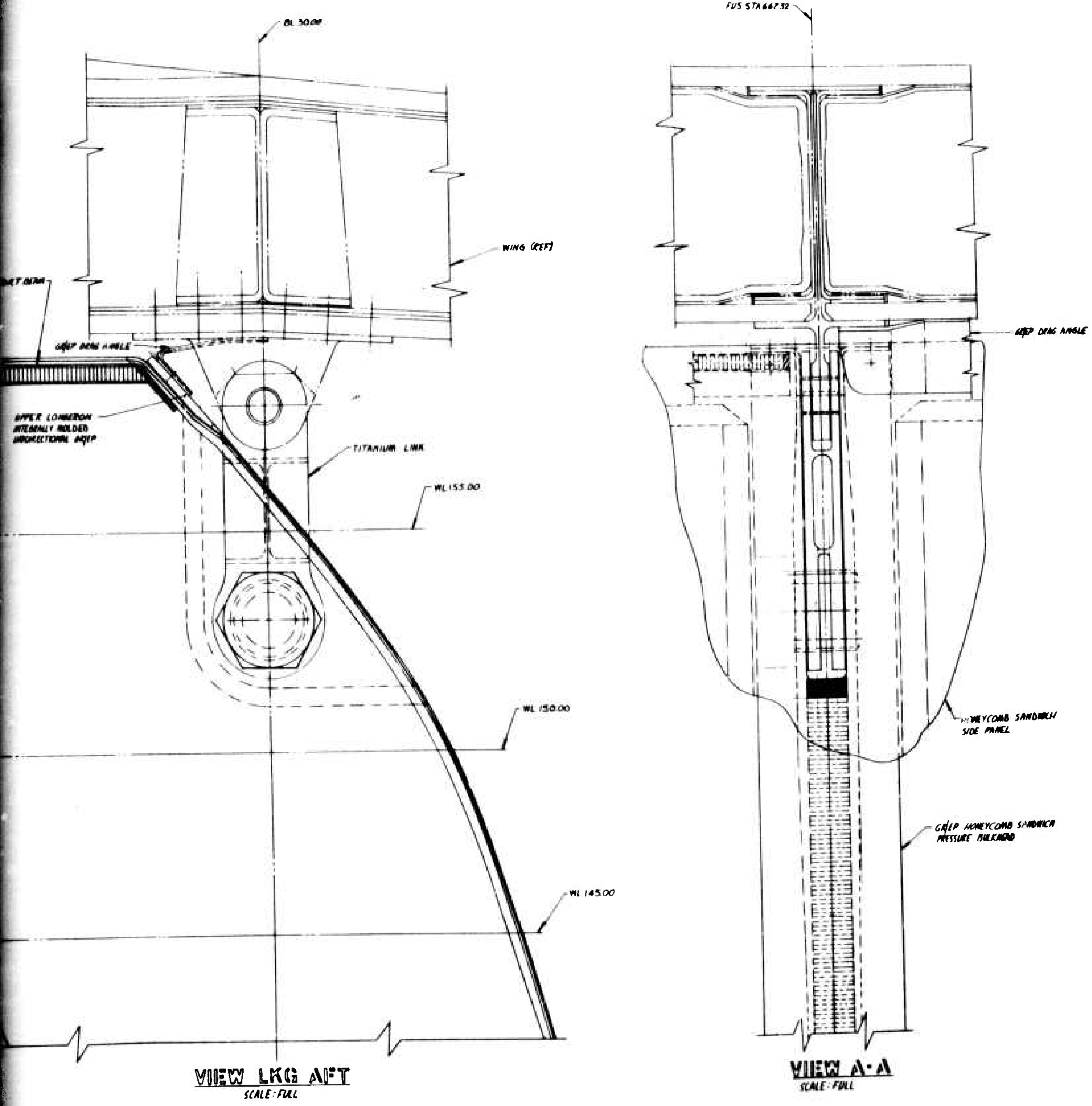
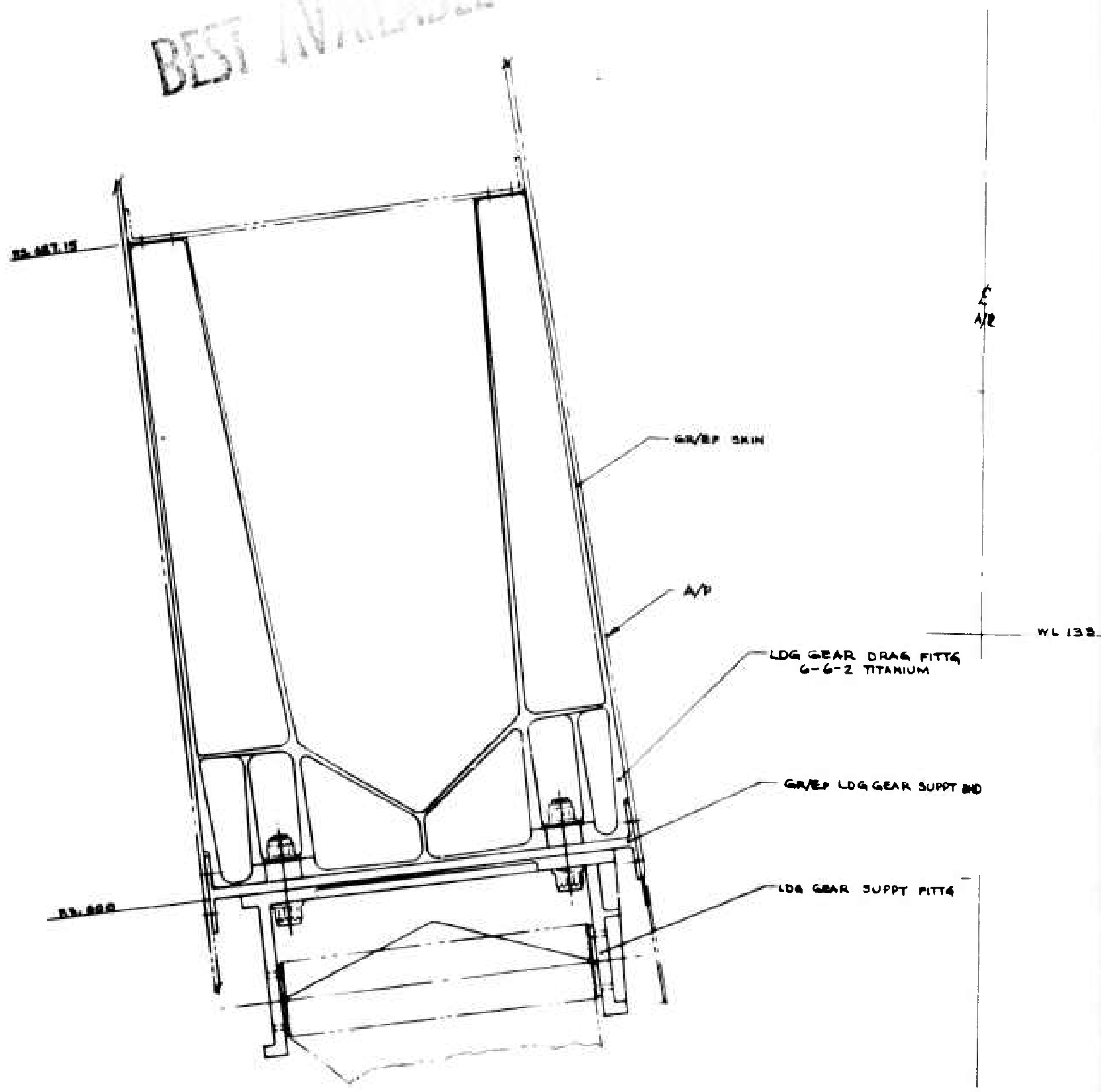


Figure 25. ADCA Wing/Fuselage Attachment (Sheet 2 of 2)



VIEW B-B
VIEW LOOKING DOWN

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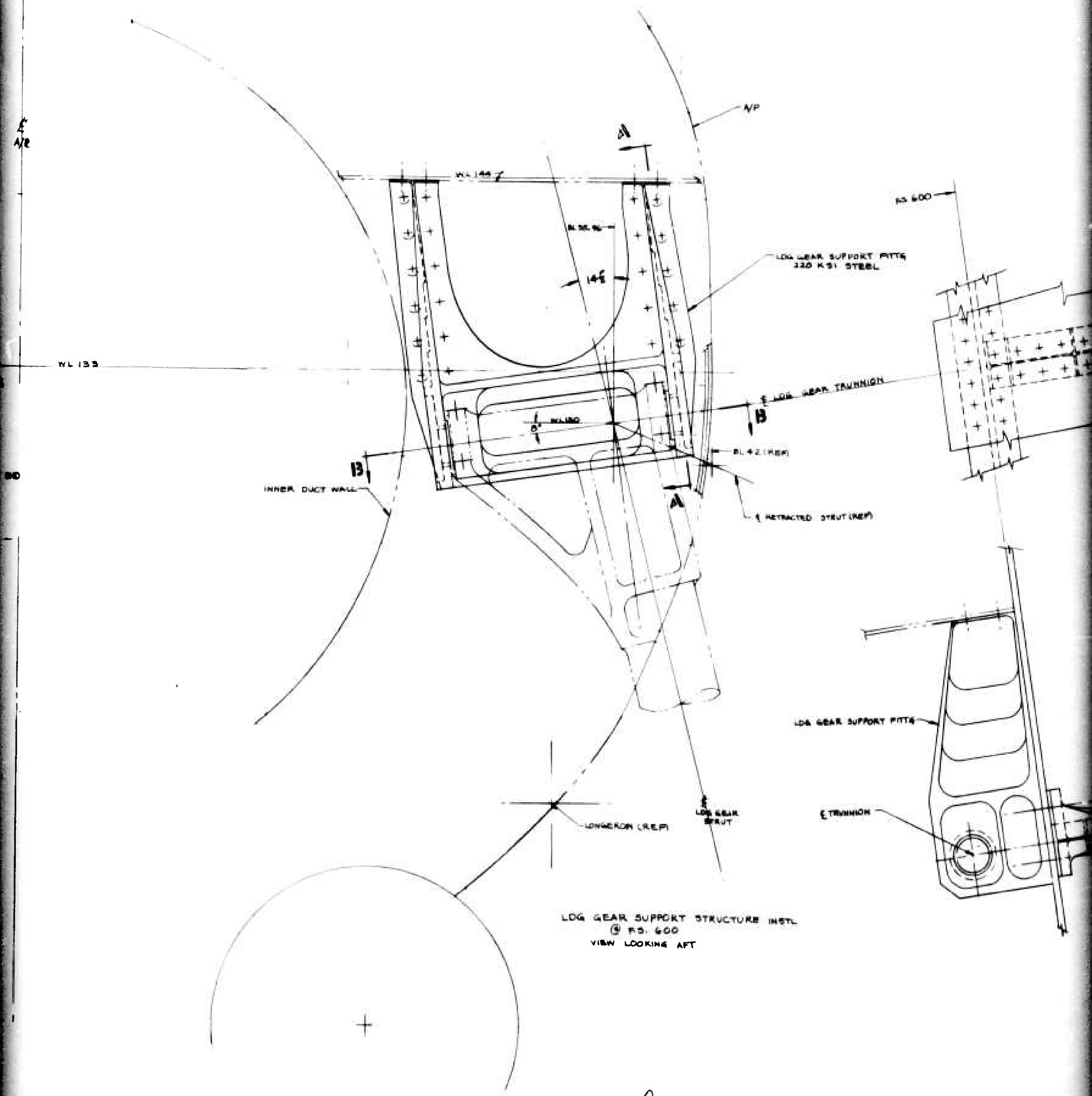

LDG GEAR SUPPORT STRUCTURE INSTL
@ F.S. 600
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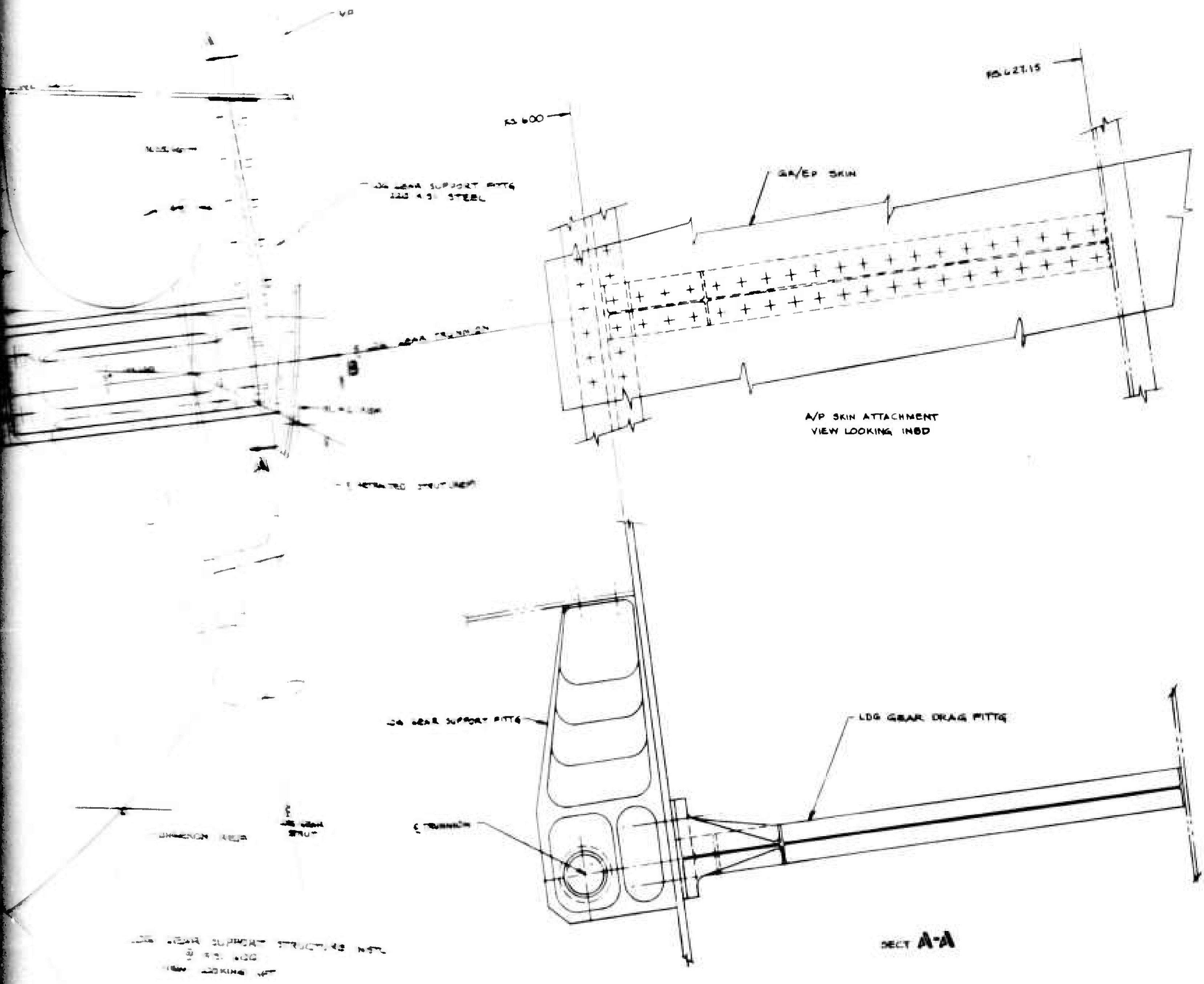


Figure 26. ADCA Main Landing Gear Support Installation

3

Section VI
FINAL ASSEMBLY

The ADCA Aircraft final assembly flow consists of joining the various components, systems installation and checkout, and final sealing/painting operations. The structural assembly task is simplified by the comparatively small number of components and their relatively high degree of completion. The wing and associated systems have been fully assembled, and the systems made fully operational before delivery to the final assembly area. The integral fuel tank portion of the mid module has been sealed and pressure-tested. The structural assembly flow consists of completing the fuselage assembly, attaching the wing, joining the vertical fin and dorsal fin, and installing the canards.

6.1 FUSELAGE ASSEMBLY

The main manufacturing splice at FS 450 joins the foreward and mid fuselage modules (Figure 27). The lower portions of the mid module side panels are attached to the forward module at FS 402 and the upper portion at FS 450. The forward module lower honeycomb skin panels are also joined to the mid module at FS 450 after the longerons are spliced.

The lower longeron of the forward module is spliced to the mid module at FS 450 by a single shear scarfed composite splice plate which is integral with the bulkhead's cap strip. The upper longeron of the forward module is spliced to a box section longeron segment in the mid module by a single shear scarfed titanium splice angle.

The joining sequence consists of bringing together the forward and mid modules with the forward module lower honeycomb skin panel being off. This facilitates the shimming of the upper longeron to the titanium splice angle, and allows the upper portion of bulkhead at FS 407 to be spliced to the lower portion which is a part of the mid module. The lower honeycomb skin panel which contains the integral forward lower longeron is installed next with the longeron being spliced. Then the forward module lower skin panels, air inlet duct cowling, and upper deck between FS 407 and FS 450 are permanently attached to complete the splice.

The tail cone is attached to the mid module at the aft engine mounting bulkhead using titanium bolts and CRES plate nuts. The main landing gear are attached to the fuselage by bolting to titanium fittings attached to bulkheads, Figure 26. The nose gear is attached to the fuselage forward keel to complete the fuselage structural assembly.

6.2 WING ATTACHMENT

The wing is lowered into place over the cavity in the mid fuselage module, and the attachment is made by joining the six titanium fittings in both the wing and fuselage, Figure 25. The drag angles and systems are then connected.

6.3 VERTICAL FIN INSTALLATION

The vertical fin to fuselage attachment is made through a titanium root fitting and splice plates, Figure 20. At the front spar, a shear pin orientated vertically attaches the root rib to a fitting attached to the frame at FS 703. After the vertical fin is installed, the dorsal fin, related fairings and rudder are attached.

6.4 CANARD INSTALLATION

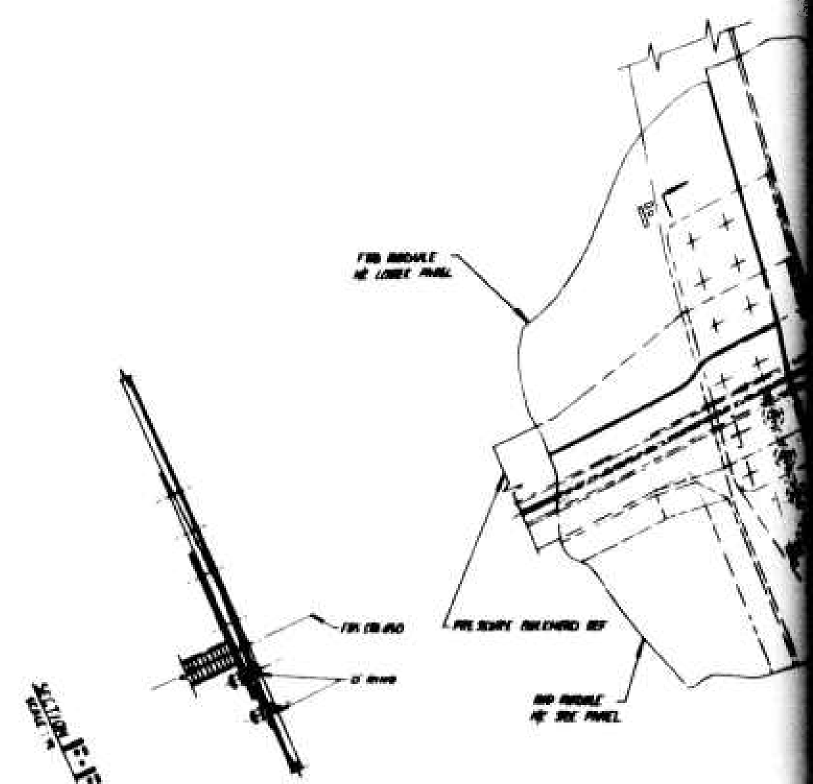
The canard spindles are inserted into the fuselage bearings and the retaining pins are installed.

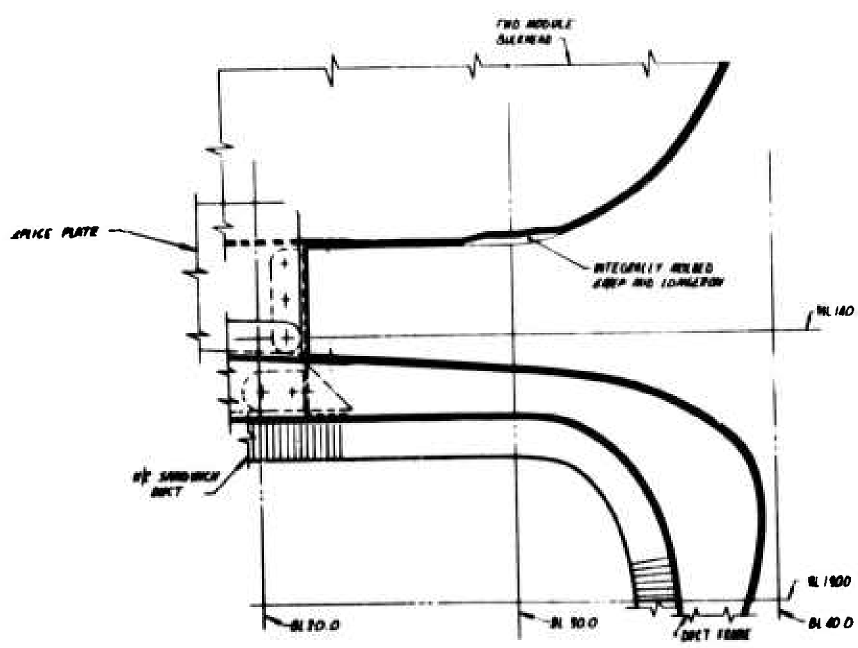
6.5 ENGINE INSTALLATION

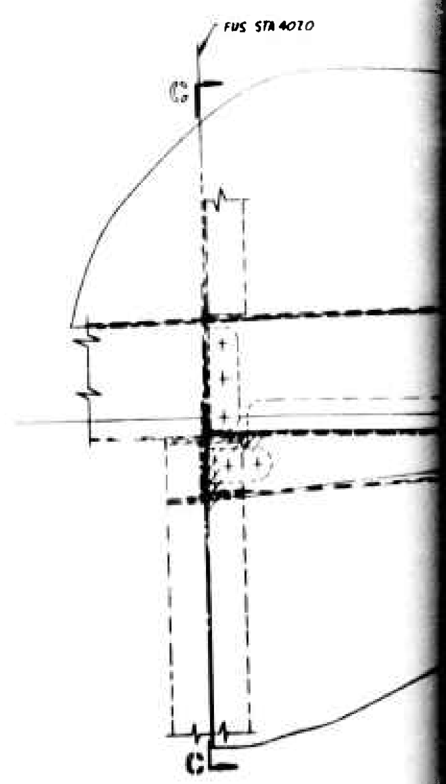
The engine is connected to titanium fittings mounted in the fuselage.

6.6 SYSTEMS INSTALLATION AND CHECKOUT

After the various assembly operations described above have been completed, installation and checkout of the various systems is completed. The access doors are then permanently attached with quick-disconnect fasteners, and the aircraft is ready for final paint.


FWD MODULE
H/C LOWER PANEL
PRESSURE BULKHEAD REF
MID MODULE
H/C SIDE PANEL
O RING
SECTION E-E


FWD MODULE
BULKHEAD
SPLICE PLATE
INTEGRALLY MOLDED
WL 140.0
H/C SANDWICH
DUCT
DUCT FRAME


FUS STA 402.0
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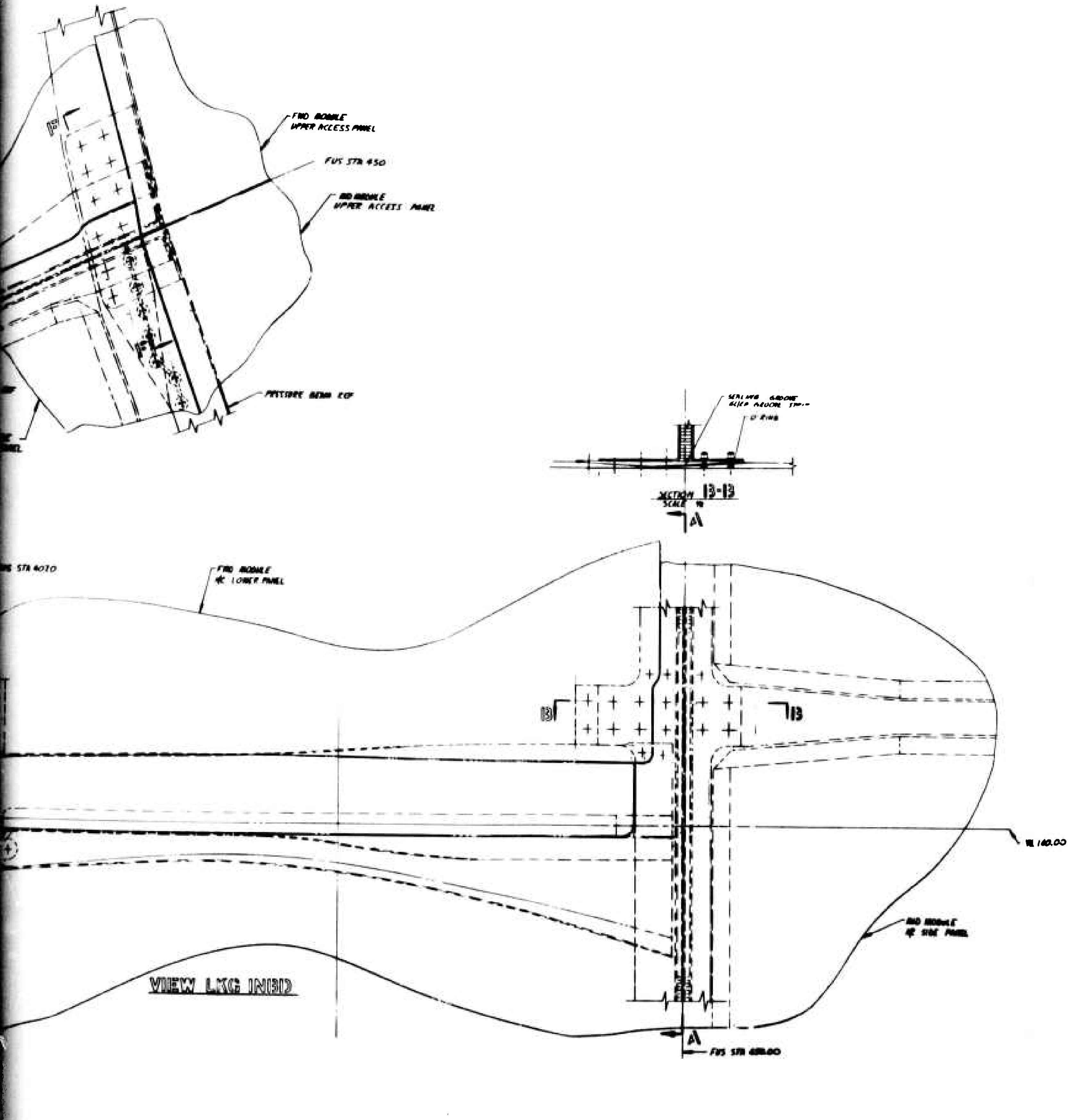
FWD MODULE
UPPER ACCESS PANEL
FUS STA 450
MID MODULE
UPPER ACCESS PANEL
PRESSURE BEAM REF
O RING
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FWD MODULE
LOWER PANEL
B
B
MID MODULE
SIDE PANEL
VIEW LKG INBD
A

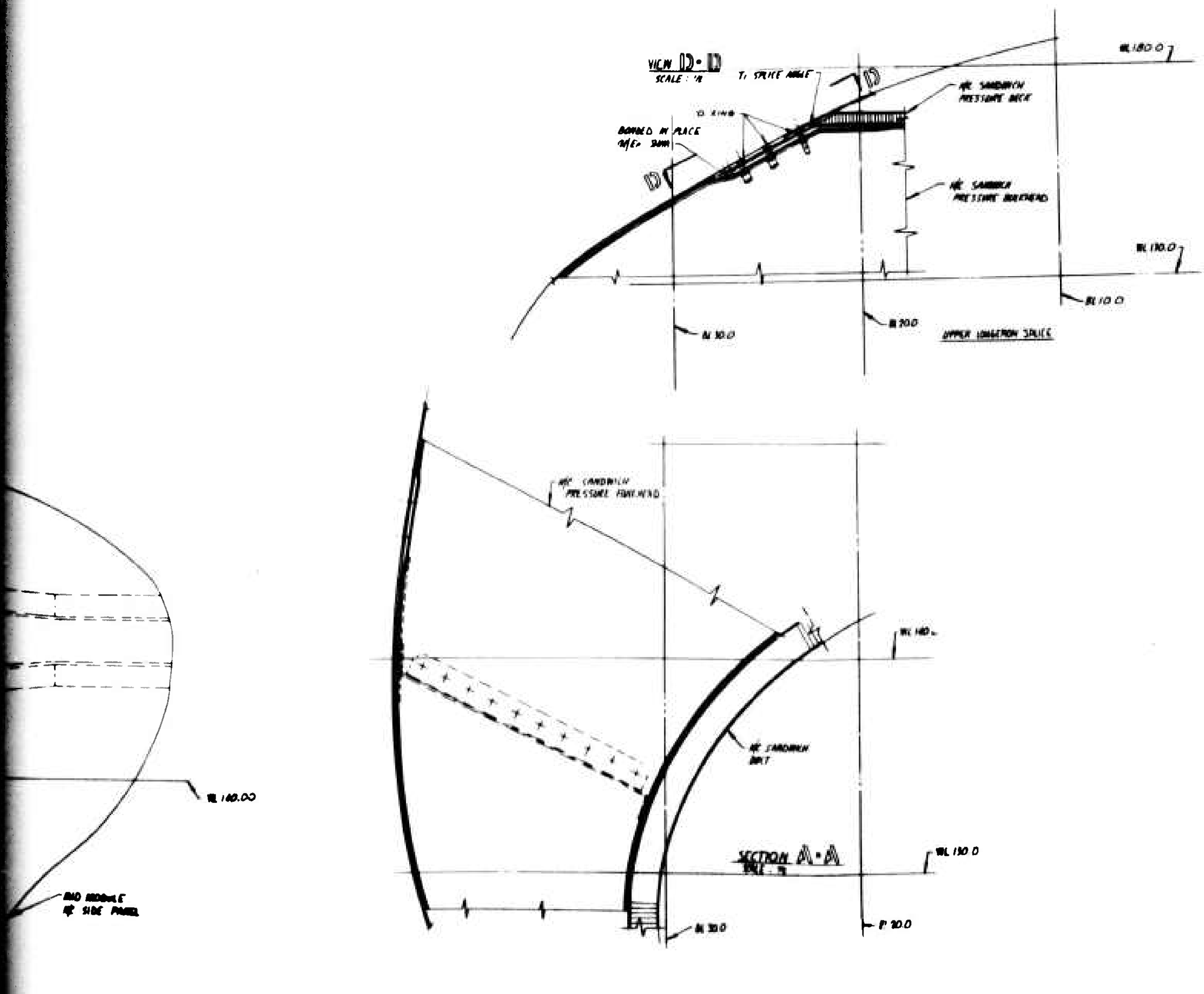

Figure 27. ADCA Manufacturing Splice Fuselage, Sta 407 and 450

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Section VII
SEALING AND FINISHING OPERATIONS

7.1 CORROSION CONTROL PLAN AND ENVIRONMENTAL SEALING

Metallic parts in contact with Graphite/Epoxy laminates are subject to galvanic corrosion. Therefore, nonmetallic materials will be used whenever possible, and the metal parts required will be fabricated from noble alloys and electrically isolated from the composite. The honeycomb materials for use in the ADCA are nonmetallic graphite, glass, or fiber/resin types such as Hexcel T-300-3K-unidirectional ±45/epoxy, HRP, HFT, and HRH-327.

The metallic fittings are generally titanium, protected with epoxy/ polyamide primer and linear polyurethane top coat. The few steel fittings used are protected by an anti-corrosive chemical conversion coating (phosphate type) and painted. Aluminum is limited to nonstructural replaceable brackets and hinges, which are sulfuric acid-anodized and sealed prior to painting. All metal fittings are isolated from the composite with .02-in. thick Gl/Ep isolation plates and nonconductive liquid shim.

The fasteners are titanium, wherever practical, or corrosion-resistant A-286 steel. Titanium fasteners installed in titanium fittings are coated with dry film lubricant to prevent fretting corrosion. All other fasteners are installed with wet sealant or similar coatings under current development. The flame-sprayed aluminum lightning strike protection system will be coated with an integrally molded .002-in. thick Gl/Ep weather seal.

The critical structural joints are sealed with polysulfide faying surface sealant covering the bolt heads and splice area. Other critical bolt heads, such as in the canard cover-to-fitting attachment, will also be sealed. All honeycomb panels are provided with Kevlar overwraps at the edges or sealed to prevent moisture incursion.

This corrosion control plan has been verified by service experience with the B/Ep F-14 Horizontal Stabilizer and humidity testing of B-1 composite Horizontal Stabilizer elements.

7.2 SURFACE FINISH

The preferred method of composite surface preparation for painting is to remove the peel ply. Where a peel ply cannot be used, the surface is scuff-sanded with 320-grit aluminum oxide. The aircraft is primed with epoxy/polyamide primer and top-coated with linear polyurethane of the specified color. The equipment bays are painted in the same manner. The cockpit interior is finished with conventional anti-reflective coatings. The interior of the fuel tank is sealed with polysulfide tank sealants and coated with a bacteria-resistant fuel tank coating (per MIL-C-27725).

Section VIII
QUALITY ASSURANCE NETWORK

Advanced composite structures currently require more quality control effort than their metallic counterparts. However, significantly less effort will be required for "production" ADCA aircraft because pertinent data is becoming increasingly available. Critical manufacturing/design requirements are being defined by production and service experience with composite structures of various types. The data to relax specification requirements, evaluate effects of defects, evaluate the serviceability of composite structures, and prepare standard repair manuals is being developed by numerous programs. When needed, these data and improved inspection techniques will be available to provide confidence in the quality assurance of an essentially all-composite aircraft.

For this program, a quality control interface was maintained from the beginning of the design effort through preparation of the manufacturing plan. This effort resulted in the establishment of a QC network for ADCA composite structure production, Figure 28.

The QC input during the design and manufacturing planning phases ensures the following:

- Elimination of designs which cannot be adequately inspected
- Highlighting of NDT procedures requiring further development
- Development of realistic but comprehensive material and fabrication specifications
- Establishment of inspection check points on the production work orders
- Definition of destruct test tabs to verify the structural integrity of associated parts
- Establishment of procedures for reporting discrepant conditions and their rework

When fabrication begins, the Quality Control Plan must be comprehensive enough to cover all operations from receiving inspection of the basic materials to final inspection of the completed component. Major items include:

- Qualification testing and source inspection at vendor's facilities as required
- Inspection of materials, parts, and assemblies purchased from outside vendors

- Surveillance of manufacturing operations and processes, including inspection of the submitted parts at predetermined points in the manufacturing cycle
- Nondestructive testing (NDT) of composite laminates and bonded assemblies
- Destructive testing of coupons to verify pretreatment of bonded metallic surfaces, response of composite materials and adhesives to molding cycles, and response of adhesives to bonding processes
- Control of assembly operations
- Documentation to report discrepant parts and their disposition
- Final acceptance of completed detail parts, subassemblics, and final assemblies

Many of these requirements can be met using existing practice for metal and fiberglass parts.

8.1 MATERIAL RECEIVING INSPECTION

In the material procurement specifications for advanced composite prepregs, the physical properties (resin and volatile eontent, taek, and resin flow) are specified, as well as mechanical properties of laminates. Fiber mechanical properties, per-ply thickness, fiber quantity per unit area, and fiber/volume ratio are also specified. As greater confidence is obtained, Quality Control can reduce the frequency of reeeiving inspection to lower program costs.

8.2 PROCESS CONTROL

The following process controls are required:

- Controlled Storage. Materials having limited shelf life must be stored in freezers when not in use and allowed to thaw in sealed bags to prevent moisture pickup prior to use. Out-time of layups must be monitored to prevent overaging prior to cure which can result in thick resin-rich laminates
- Control of Layup Areas. Layup areas must be environmentally controlled to prevent structurally harmful moisture pickup or prepreg handling problems due to temperature variations. General housekeeping of the area and the use of solvents and mold releases within the layup room must be monitored
- Tool Inspection. All tools used during the program will be initially inspected to ensure conformance to tool designs, and reinspected periodically

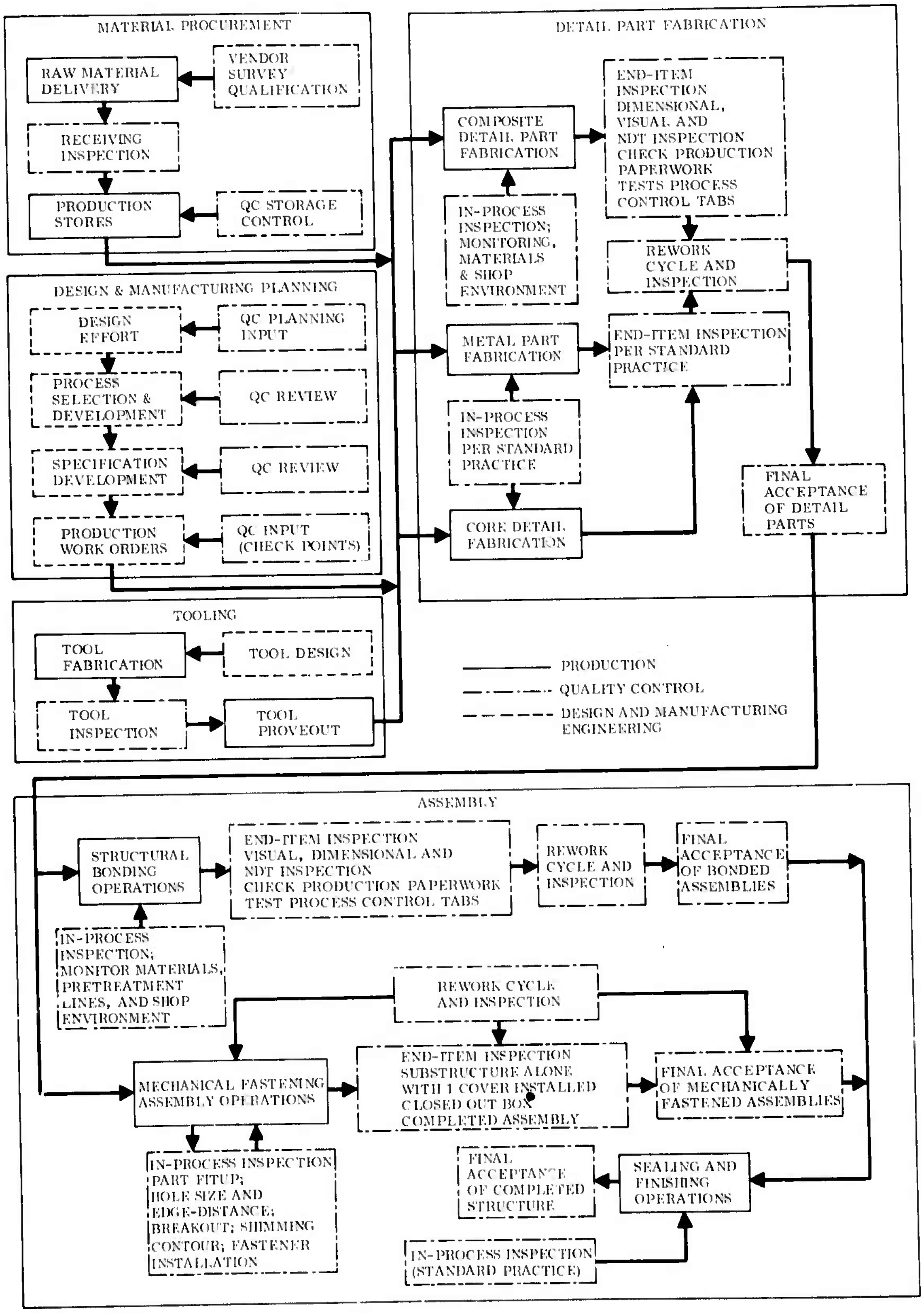

Figure 28. QC Network for ADCA Composite Production Programs

- Molded Parts. Composite laminates will be monitored by Quality Control through each phase of manufacture:
  - Layup. Individual plies for each assembly will be inspected for the following characteristics after the ply is laid-up on the transparent template or on the tool:
    (1) Fiber orientation
    (2) No gaps larger than specified, overlaps, or contaminants
    (3) Position of scrim cloth (up or down)
    (4) Ply trim
    (5) Proper sequence of stacking on the mold form
    (6) Adhesive layup and surface preparation of cocured core, metal, or fiberglass detail parts
  - Cure. The following controls will be applied for the autoclave cure cycle:
    (1) Inspection of the nylon pressure bag
    (2) Review of time, temperature, pressure and vacuum charts, and heat-up and cooldown rates
    (3) Presence of and proper identification of the appropriate process control test panels

    The oven post-cure will be inspected in the same manner
  - Dimensions. Laminate thickness measurements will be made to determine per ply thickness using eddy current or mechanical means. Peripheral measurement will be verified on completion of the triming operation or as molded for net trim parts
  - Laminate Process Control (PC) Test Panels. PC test panels will be representative of the parts being cured to verify the response of the materials to the cure cycle. Process control panels typically include 15-ply laminates for skins, cocured sandwich panels, and lap shears for secondary bonding operations

8.3 END ITEM NDT INSPECTION

Significant advances in NDT inspection make possible the detection of structurally harmful internal flaws in detail parts and assemblies. The flaws shown in Figure 29. can be uncovered using a combination of ultrasonic and radiographic techniques.

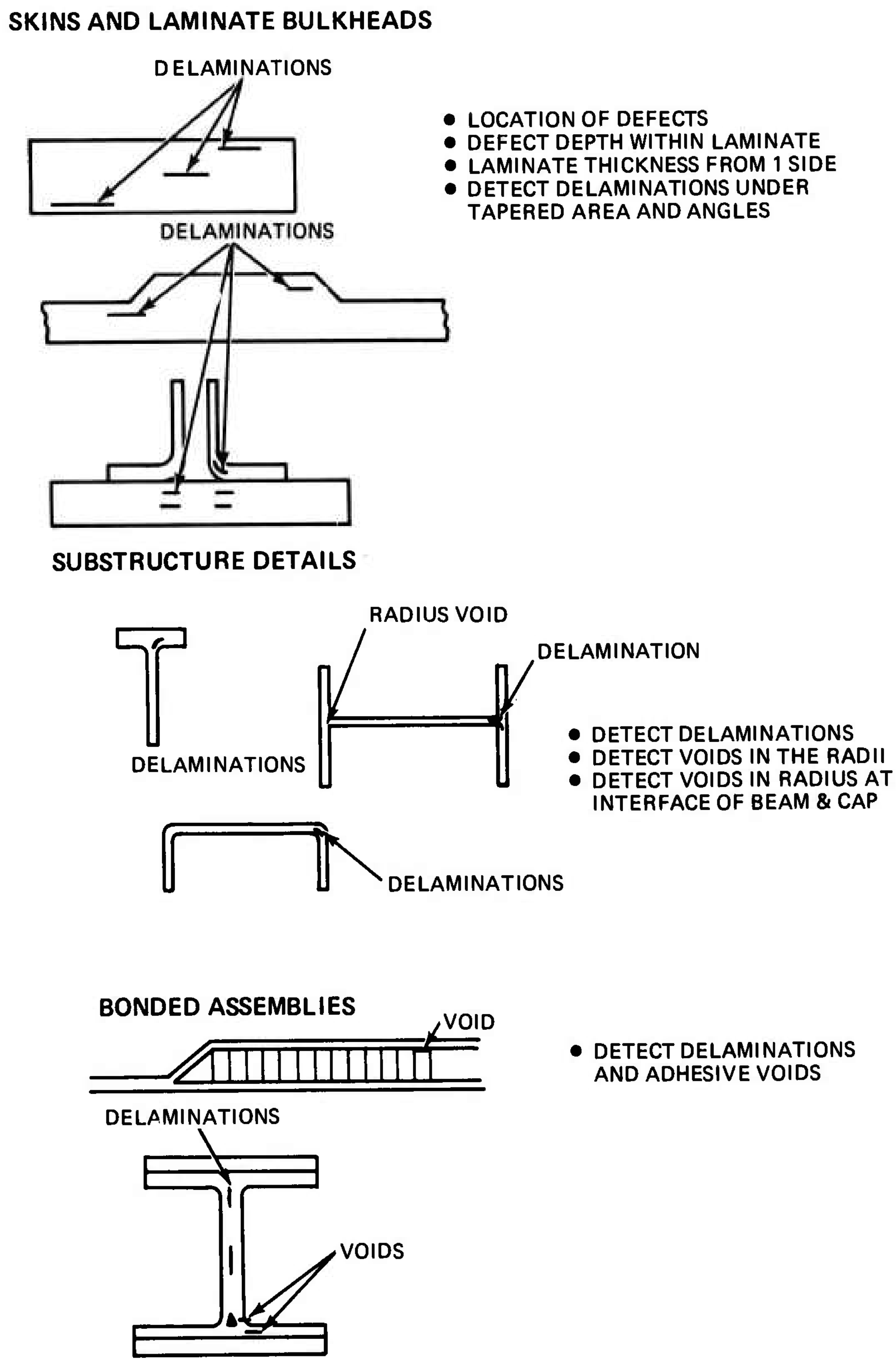

Figure 29. Structural Defects Detectable by Nondestructive Test

Ultrasonic inspection can be performed using one of three basic techniques: pulse-echo, through transmission, and resonance. The pulse echo technique is applicable to solid laminate detail parts, such as skins and bulkheads which can be immersed. Through transmission ultrasonics is applicable to bonded structure. Resonance ultrasonics can be used to reinspect local areas to verify potential defects found by the other methods. Also, with appropriate standards, resonance ultrasonics can accurately predict the depth of a delamination, thereby assisting in planning repairs.

Radiography has been demonstrated to be an effective method of determining radius defects using x-rays directed parallel to the width of the delamination (Figure 30). In the radius areas of channel sections and spars, the X-ray beam directed at a 45° angle to the radius can uncover voids due to lack of adhesive or improper ply compaction. Available radiographic standards are used to characterize the defect. The radiographic process parameters are: 60 Kv power, 8 ma/min exposure, 36 in. source-to-film distance, GAF-100 film or equivalent, and X-O mat film processing.

8.4 ASSEMBLY INSPECTION

Manufacturing inspection requirements for composite fastened assemblies have some similarities to those for metal structure. These include hole size and location, fastener installation standards, contour control, sealing requirements, good workmanship, and electrical grounding. Also, composites place three additional requirements on floor inspection. Breakout produced by defective drilling procedures must be minimized by constant surveillance. Then, the fit up of parts prior to fastening must be carefully checked to prevent internal damage. Typically, gaps larger than .005 in. at the fastener should be liquid-shimmed. Liquid shim accommodates gaps of .005 to 0.03 in. Larger gaps require use of both solid and liquid shim. Lastly, floor inspection must ensure that the corrosion control plan is properly executed. Verification of wet installation of fasteners, coatings on fittings, use of specified noble metal fasteners, and environmental sealing must be obtained.

8.5 MRR ACTION AND ALLOWABLE REPAIRS

A control system for dispositioning and controlling all nonconforming materials, detail parts, and components must be established. The system must provide for reporting the discrepancy, stopping further processing until a disposition is made, documenting the disposition and any corrective action required, and providing traceability at a future date.

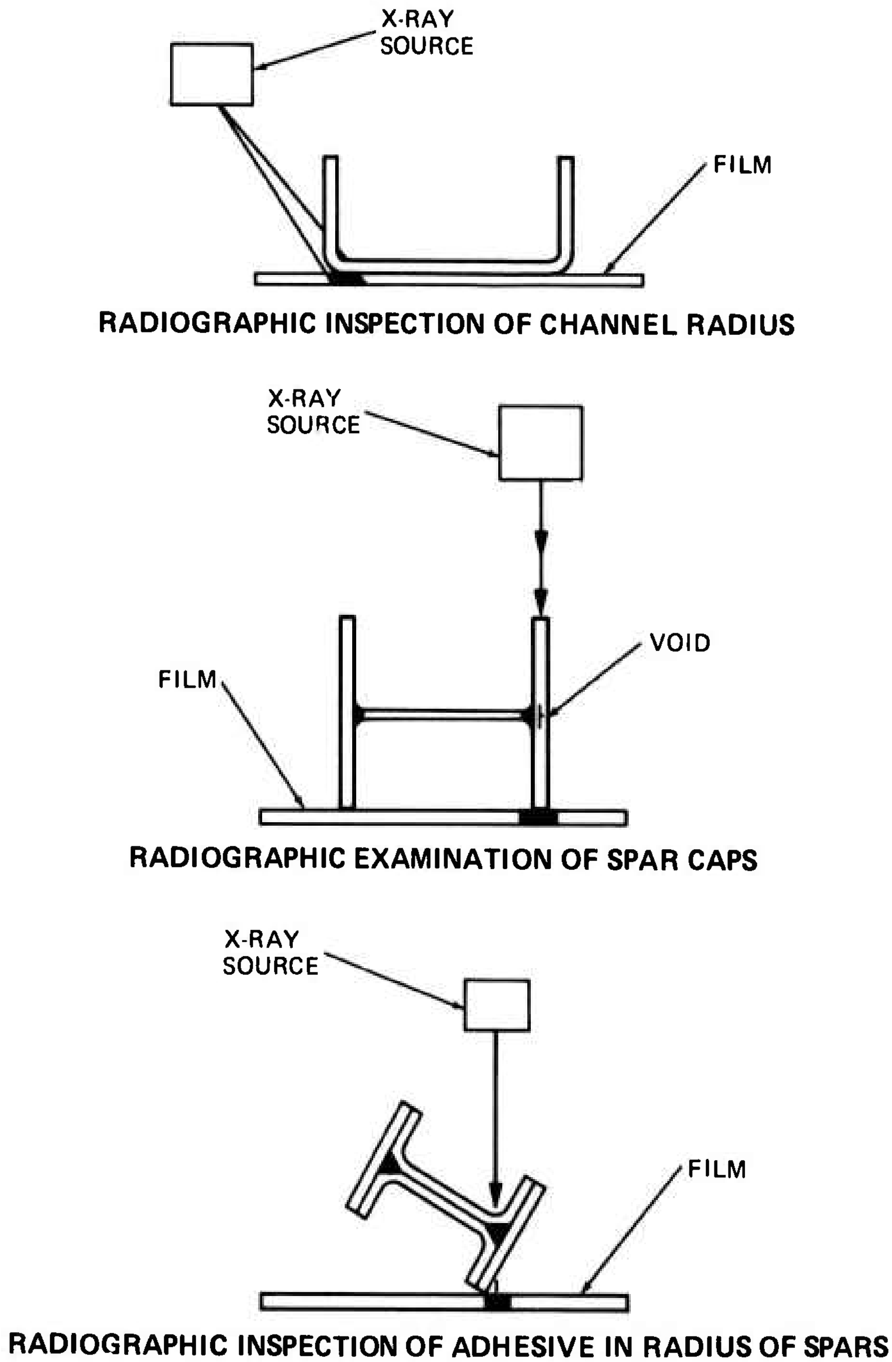

Figure 30. Radiographic Inspection

Based on engineering data, a standard repair manual will provide a listing and description of commonly occurring minor discrepancies and the standard repair procedure for each. This manual will permit a standardized recording and reporting method and assure proper implementation of the applicable repair procedure. Repairs discussed in the manual will be general in nature and may be used singly or in combination as required.

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